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**ARMY PRELIMINARY EVALUATION JOH-58A HELICOPTER WITH  
LOW REFLECTIVE PAINT AND INFRARED COUNTERMEASURE  
EXHAUST SYSTEM**

**ARMY AVIATION ENGINEERING FLIGHT ACTIVITY**

**DECEMBER 1975**

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USAAEFA PROJECT NO. 75-11



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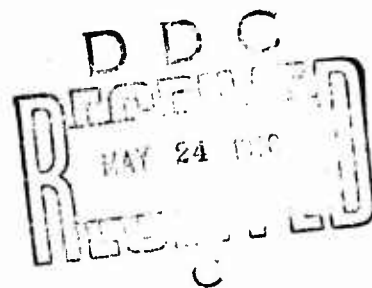
FINAL REPORT

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DECEMBER 1975



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UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY  
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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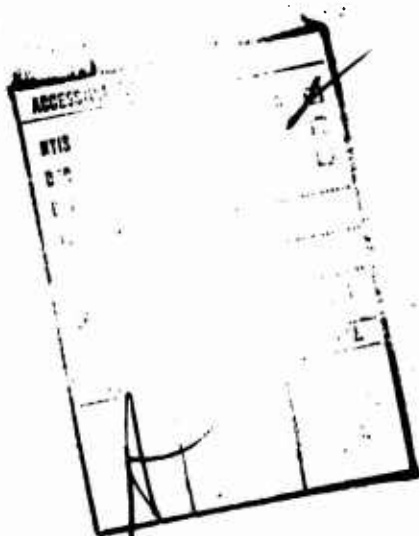


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**20. Abstract**

and handling qualities characteristics and with preliminary base-line testing performed during this evaluation. One deficiency and two shortcomings were noted during this evaluation. Although not classified as a deficiency or shortcoming, the most significant finding in this evaluation was the serious degradation in hover and level flight performance when the rotor blades are painted with low reflective paint.



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# **INTRODUCTION**

## **BACKGROUND**

1. The Aircraft Survivability Equipment (ASE) Product Manager is currently attempting to provide the OH-58A helicopter with increased survivability from the enemy infrared (IR) and heat-seeking missile threat. The United States Army Aviation Engineering Flight Activity (USAAEFA) was tasked by the United States Army Aviation Systems Command (AVSCOM) (ref 1, app A) to evaluate the effect of a low reflective (LR) paint and an IR countermeasure (IRCM) exhaust system on the performance and handling qualities of an OH-58A helicopter.

## **TEST OBJECTIVES**

2. The test objectives were as follows:

a. To verify the airworthiness of the helicopter with LR paint and the IRCM exhaust stacks installed.

b. To determine the changes in the OH-58A handling qualities and performance characteristics with LR paint and the IRCM exhaust stacks installed.

## **DESCRIPTION**

3. The OH-58A is a single-main-rotor turbine-powered light observation helicopter manufactured by Bell Helicopter Company (BHC), Fort Worth, Texas. The main rotor is a two-bladed, semirigid, teetering type and the tail rotor is a two-bladed, semirigid, delta-three hinge type. The cockpit configuration is two-place (pilot and copilot/observer) and the cargo compartment has provisions for two passengers. All primary flight controls are hydraulically boosted and irreversible, while the directional control on the standard OH-58A aircraft is unboosted and reversible. The main landing gear is a fixed, energy-absorbing skid type. The helicopter is powered by an Allison T63-A-700 free gas turbine engine with a takeoff power rating of 317 shaft horsepower (shp) under sea-level, standard-day uninstalled conditions. The main transmission has a rating of 270 shp (maximum continuous) with a takeoff power limit of 317 shp (5-minute rating). A more detailed description of the OH-58A helicopter is contained in the operator's manual (ref 2, app A). The modifications incorporated in the test helicopter which resulted in the "J" designation were installation of a BHC electronic 3-axis stability and control augmentation system (SCAS) which incorporates a hydraulically boosted and irreversible directional control and a Sperry helicopter command information system (HCIS). The SCAS was not used during handling qualities testing. A more

detailed description of the SCAS and HCIS is contained in reference 3. The aircraft was further modified by the installation of IRCM exhaust stacks and application of LR paint, as described in paragraphs 4 and 5, respectively.

4. The prototype exhaust stacks used in this evaluation were designed to reduce the IR signature of the aircraft. They were designed to use the exhaust nozzle as an aspirating pump to draw cool air through an annular opening in the stack. The cooling air helps to protect the shield section of the stack from hot exhaust gases. The shield is also cooled by the flow of ambient air over the cooling fins. The exhaust stacks were manufactured by the Hughes Helicopter Division of Summa Corporation, Culver City, California. Photographs of the exhaust stacks are shown in appendix B. A more detailed description is contained in appendix C.

5. The paint used on the test aircraft is a low reflective olive drab acrylic lacquer (FSN 8910-083-6588) specified by military specification MIL-L-46159. It is formulated to reduce IR solar reflections in the spectral band pass of all currently identified IR-seeking missiles. It is also designed to have a low visual gloss to aid in visual contrast reduction. The exterior surfaces were painted with the LR paint with the exception of high conspicuity and/or safety markings, main rotor mast, tail rotor drive, tail rotor blades, and main rotor blade leading edges (ref 10, app A). Painting of the main rotor blades in accordance with reference 10 resulted in the rotor blade having 76 percent of chord (measured from the trailing edge) covered with the LR paint.

#### TEST SCOPE

6. This APE provided a limited evaluation of performance, handling qualities, and engine characteristics of the JOH-58A helicopter equipped with LR paint and IRCM exhaust stacks. The configurations tested are describe in paragraph 9 and table 1. Testing was performed at Edwards Air Force Base (2302 feet field elevation) and at the Pacific Missile Test Center (13 feet field elevation), Point Mugu, California, from 1 July through 14 October 1975. Forty-eight test flights for a total of 41 productive flight test hours were flown. The test aircraft was evaluated against the requirements of military specification MIL-H-8501A (ref 4, app A), with the deviations contained in reference 12.

7. Test results were compared to those previously reported for the OH-58A helicopter with standard paint and exhaust stacks (refs 5 and 6, app A). Flight limitations contained in the operator's manual and the safety-of-flight release (ref 7) were observed during the tests, with the exception of minimum transient rotor speed which was violated inadvertently during the simulated sudden engine failure tests.

## **TEST METHODOLOGY**

8. Standard flight test techniques were used for the handling qualities and performance testing (refs 8, 9, and 13, app A). Test methods are described briefly in the Results and Discussion section of this report. A detailed description of the test instrumentation is contained in appendix F. A description of the data reduction procedures is contained in appendix E. Pilot comments were used to aid in the analysis of data and to determine the overall qualitative assessment of the flying qualities of the JOH-58A helicopter with LR paint and IRCM exhaust system installed. The Handling Qualities Rating Scale (HQRS) used to augment pilot qualitative comments is included as figure 1, appendix E.

## **AIRCRAFT TEST CONFIGURATIONS**

9. Initial testing for base-line data was performed with the test helicopter in the standard configuration (configuration A). After the initial testing had been completed, the aircraft was equipped with the IRCM exhaust stacks (configuration B) and 2.7 hours of testing were completed. After testing in configuration B, the helicopter was delivered to Sharpe Army Depot, California, where it was painted with LR paint (ref 10, app A). The main rotor blades were painted the entire length of the blade from the trailing edge forward to the point of maximum airfoil thickness (configuration C). Nineteen hours of testing were completed in this configuration. Additional testing was then conducted with the same paint as configuration C but with standard exhaust stacks (configuration D). After testing with configurations A, B, C, and D, it was thought that the small ridge of paint which existed where the normal acrylic lacquer, used from the leading edge to a point of maximum airfoil thickness met the LR paint might be a significant performance factor. The ridge was removed by sanding the area of transition between the normal acrylic lacquer on the leading edge and the LR paint on the remainder of the rotor blade (configuration E) and further testing was conducted. Additional testing was accomplished with the IRCM exhaust stacks and a new set of rotor blades with standard lacquer paint (configuration F) in an attempt to determine the amount of degradation in performance attributable to the painted rotor blades. The remaining tests were conducted with standard exhaust stacks and standard rotor blades (configuration G). An abbreviated description of the aircraft configurations is contained in table 1.

Table 1. Aircraft Test Configurations.

Configuration Designation	Configuration Description
A	Std paint, std stacks, std rotors (basic aircraft)
B	Std paint, IRCM stacks <sup>1</sup> , std rotors
C	LR paint <sup>2</sup> , IRCM stacks, LR rotors <sup>3</sup>
D	LR paint, std stacks, LR rotors
E	LR paint, IRCM stacks, LR rotor (ridge removed) <sup>4</sup>
F	LR paint, IRCM stacks, std rotor
G	LR paint, std stacks, std rotor

<sup>1</sup>IRCM stacks: Hughes IRCM exhaust suppressor stacks installed.

<sup>2</sup>LR paint: Fuselage painted with LR acrylic lacquer.

<sup>3</sup>LR rotors: Rotor blades painted with LR acrylic lacquer.

<sup>4</sup>LR rotor (ridge removed): LR rotor that was sanded to remove the paint ridge formed between paint applied to leading edge and the LR paint on the remainder of the rotor blade.

# RESULTS AND DISCUSSION

## GENERAL

10. An APE of the OH-58A helicopter was performed to determine any changes in the OH-58A performance and handling qualities characteristics when painted with LR paint and equipped with IRCM exhaust stacks. Results of these tests were compared with results from earlier evaluations of the OH-58A performance and handling qualities characteristics and with preliminary base-line testing performed during this evaluation. One deficiency and two shortcomings were noted. Although not classified as a deficiency or shortcoming, the most significant finding in this evaluation was the serious degradation in hover and level flight performance when the rotor blades are painted with LR paint.

## PERFORMANCE

### Hover

11. Hover performance capabilities of the OH-58A helicopter were evaluated in ground effect (IGE) and out of ground effect (OGE) at various skid heights under the conditions shown in table 2. The free flight hover method was used to determine hover performance capability. Skid height was measured by visual reference to a measured weighted cord attached to the left skid. Incremental amounts of ballast were added to the helicopter until the gross weight was such that either the engine temperature or transmission power limit was reached.

12. The hover performance capabilities are summarized and presented in figures 1 through 11, appendix G. Nondimensional plots of power coefficient ( $C_p$ ) versus thrust coefficient ( $C_T$ ) are presented for configuration C (LR-painted fuselage and rotor and IRCM exhaust stacks) in figures 1 through 7. A comparison of the standard-day and 35°C hot-day 4-foot IGE and OGE hover ceiling is presented in figures A and B for configurations A and C. Depicted in table 3 is the loss in gross weight hover capability incurred with LR paint and/or installation of the IRCM exhaust stacks.

13. The standard-day OGE hover ceiling decreased from 5100 feet for the standard aircraft (configuration A) to 2850 feet for configuration C at 3000 pounds gross weight (fig. 8). The 35°C hot-day 4-foot IGE hover ceiling decreased from 2200 feet (configuration A) to a point that the aircraft will only hover at 2980 pounds gross weight at sea level, as shown in figure A.



Table 2. Hover Performance.<sup>1</sup>

Configuration	Skid Height <sup>2,3</sup> (ft)	Gross Weight (lb)	Density Altitude (ft)	Outside Air Temperature (°C)
A	2, 4, 10, 20, and 50	2750 to 3250	4040	24.5
B	Not tested			
C	2, 4, 10, 20, and 50	2315 to 3500	Sea level to 4150	15.5 to 27.0
D	50	2450 to 3010	340	15.5
E	50	2530 to 3040	2860	16.0
F	50	2450 to 3110	2820	15.5
G	2 and 50	2090 to 3030	2820	14.5

<sup>1</sup>Rotor speed: 347 to 354 rpm.

<sup>2</sup>Free flight hover technique used.

<sup>3</sup>2, 4, 10, and 20 feet are IGE hover points, and 50 feet is OGE hover.

Table 3. OGE Hover Weight Capability.<sup>1</sup>

Configuration	Gross Weight (lb)	Decrease in Lift Capability (lb)
A	2930	---
C	2680	250
D	2790	140
E	2740	190
F	2820	110
G	2930	---

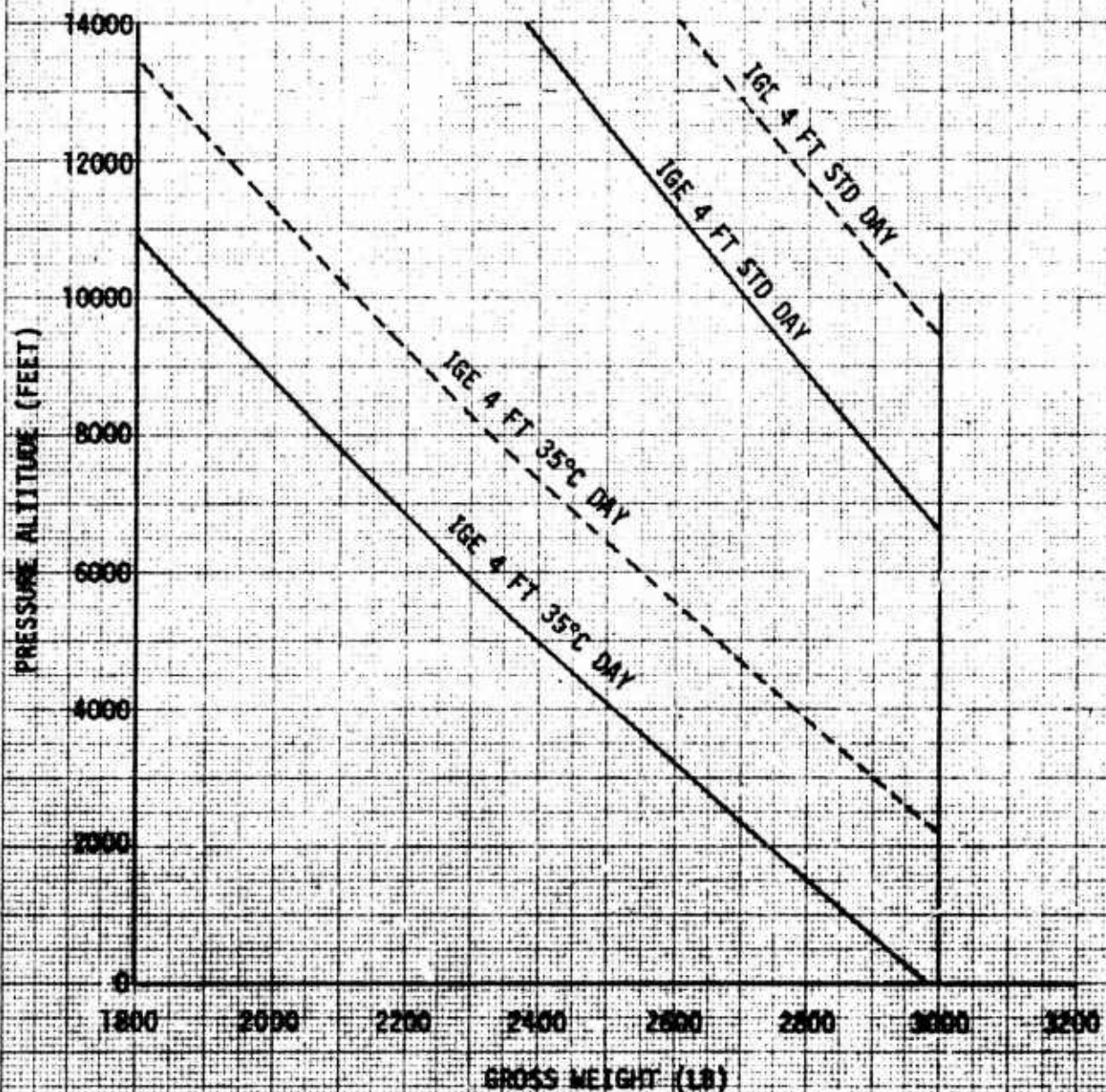
<sup>1</sup>Sea level, 35°C hot day.

**FIGURE A**  
**SUMMARY HOVER PERFORMANCE COMPARISON**  
**UH-58A**  
**S/N USA 68-16706**  
**FREE FLIGHT METHOD**  
**WINDS LESS THAN 3 KNOTS**  
**HOVER IN 4 FOOT GROUND EFFECT**

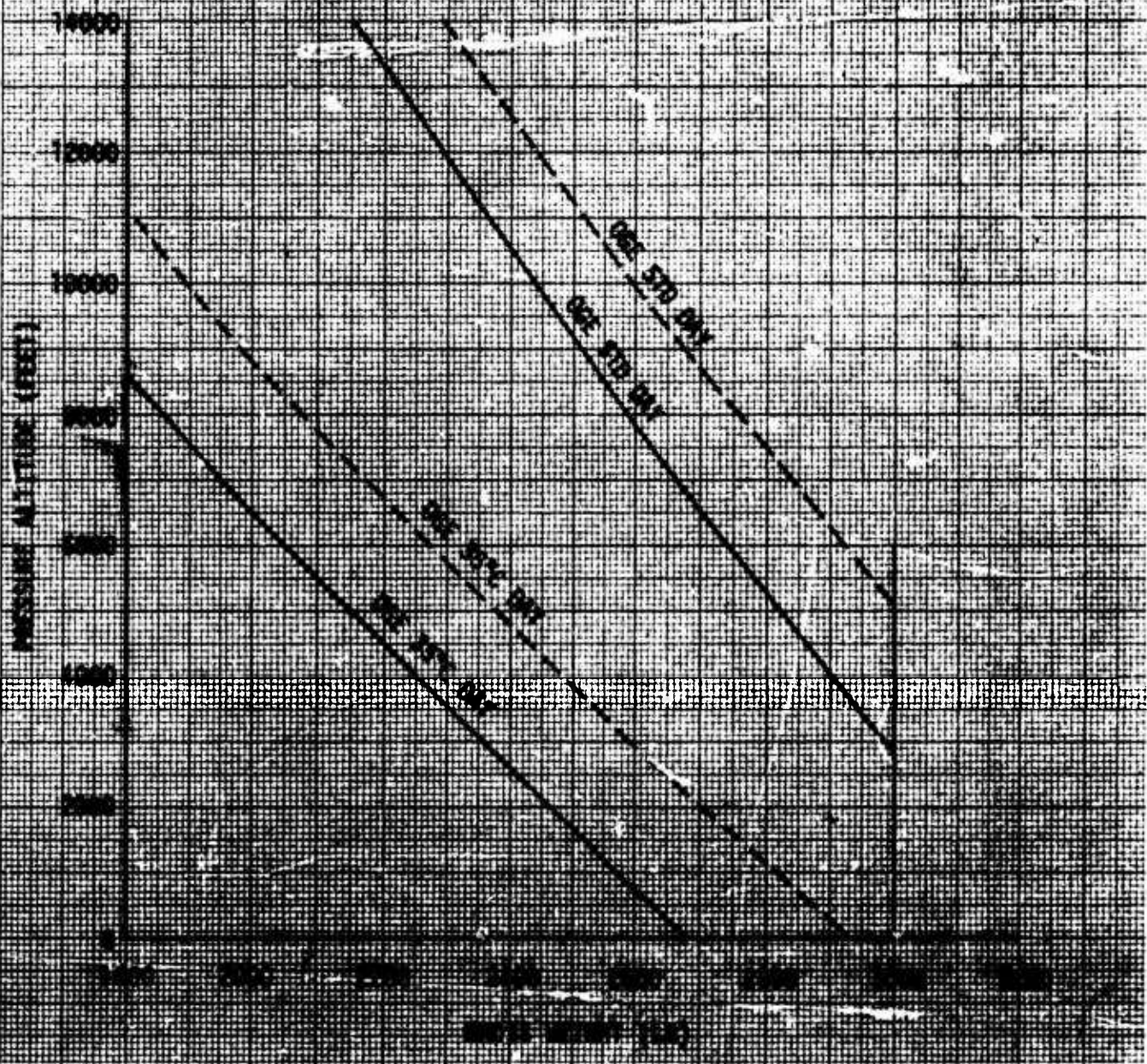
**CONFIGURATION**

----- A  
 \_\_\_\_\_ C

- NOTE:** 1 Configuration A denotes std paint, std stacks, std rotor  
 2 Configuration C denotes LR paint, LR stacks, LR rotor  
 3 Maximum gross weight limit of 3000 pounds from operator's manual for OH-58A helicopter



NOTE: This chart is intended for use with the standard atmosphere. It is not valid for use with other atmospheres. It is not valid for use with other than standard sea level pressure of 30.0 inches Hg (1013.25 mb).





14. With the paint ridge removed (configuration E), hover performance was slightly improved over configuration C, as shown in figure 9, appendix G. The OGE hover ceiling for configuration E was 3600 feet, standard day, at 3000 pounds gross weight and for a 35°C hot day, maximum gross weight for sea-level hover (OGE) was 2740 pounds, as computed from figures 9 and 60.

15. As depicted in figure 11, appendix G, the OH-58A hover performance is not degraded by applying LR paint to the fuselage. However, painting the rotor blades with LR paint greatly reduced the capability of the OH-58A helicopter to perform mission maneuvers, in that OGE hover lift capability was reduced approximately 5 percent.

16. As shown in figures 8 and 10, appendix G, the OH-58A hover capability was significantly reduced by the addition of the IRCM exhaust stacks. As depicted, the addition of the stacks resulted in an approximate 4 percent loss in lift capability in an OGE hover.

17. The engine inlet air temperature ( $T_2$ ) was monitored throughout the hover evaluation and within the scope of this test, the aircraft did not ingest exhaust gases.

#### Level Flight

18. Tests were conducted to determine airspeed, fuel flow, and power-required relationships to define the OH-58A level flight performance. Conditions and configurations tested are presented in table 4. All flights were conducted at zero sideslip in stabilized unaccelerated level flight. Each speed power test was conducted at a constant weight-to-density ratio ( $W/\sigma$ ). This procedure necessitated an increase in altitude for successive data points as fuel was consumed.

19. Test results, to include nondimensional summary plots, specific range, various individual configuration power polars, and endurance summaries, are presented in figures 12 through 26, appendix G. The nondimensional summary plots, specific range, endurance summaries, and the individual power polars are presented in figures 12 through 23 for configuration F. Test results are summarized in table 5.

20. The standard aircraft with IRCM exhaust stacks installed (configuration B) had an increase in parasite drag of 2.0 square feet of equivalent flat plate area ( $\Delta f_e$ ) over the standard aircraft (configuration A), as shown in figure 24, appendix G. An increase of 18 shp was required at 100 knots true airspeed (KTAS), which was attributed to the increase in  $f_e$  and upward thrust component of the engine exhaust.

Table 4. Level Flight Performance.<sup>1</sup>

Configuration	Average Gross Weight (lb)	Average Density Altitude (ft)	Outside Air Temperature (°C)	Thrust Coefficient	True Airspeed Range (kt)
A	2820	7200	17.0	0.003498	35 to 113
B	2820	7240	24.0	0.003503	35 to 105
C	3780	7880	18.0	0.003522	35 to 102
	2520	4080	27.0	0.002843	35 to 109
D	No flights in this configuration				
E	2460	4420	19.5	0.002804	34 to 115
	2860	10,940	10.5	0.003987	32 to 102
	2820	5240	20.5	0.003295	24 to 110
F	2620	9620	12.0	0.003515	36 to 110
	2880	9100	12.5	0.003790	25 to 102
	2560	3320	24.0	0.002815	23 to 115

<sup>1</sup>Rotor speed for all tests: 354 rpm.  
Bleed air OFF, all doors ON, unarmed.

Table 5. Level Flight Performance Comparison.<sup>1</sup>

Configuration	Power Required at 100 KTAS (shp)	Increase in Power Required Over Standard Aircraft ( $\Delta$ shp)	Specific Range (NAMPP)	Reduction in Specific Range From Configuration A (%)
A	206	—	0.625	—
C	244	38	0.541	13
E	231	25	0.568	9
F	223	17	0.588	6

<sup>1</sup>At 2800 pounds gross weight under standard-day, sea-level conditions.

21. Configuration C (LR-painted fuselage and rotor and IRCM exhaust stacks) had a large increase in power required over configuration A. As depicted in figure 25, appendix G, 20 more shp (9 percent of available) was required for 100 KTAS for configuration C. The large increase in power required over configuration A can be attributed to the increase in  $f_e$  due to the IRCM exhaust stacks and an increase in induced and profile drag from the LR paint.

22. As indicated in figure 26, appendix G, configuration E (LR-painted fuselage, modified LR-painted rotor, and IRCM exhaust stacks) showed an improvement in level flight performance over configuration C but still required a large power increase over the standard aircraft. For 100 KTAS an increase of 19 shp was required for configuration E, also attributed to the increase in  $f_e$  and the LR paint.

23. Configuration F (LR-painted fuselage, standard rotor, and IRCM exhaust stacks) (figs. 19 through 23, app G; summarized in figs. 12 through 14) shows some decrease in power required when compared to configurations D and E; however, a significant increase still exists in power required over the standard aircraft. The increase in power required can be attributed to the parasitic drag increase of the IRCM exhaust stacks and the LR-painted fuselage.

Configuration F decreased the specific range by 14 percent under standard-day, sea-level conditions from that of the standard aircraft, as shown in figure 15.

24. Level flight performance was degraded by the addition of LR paint and/or IRCM exhaust stacks. Painting the rotor blades with LR paint degraded the level flight performance to an extent that the aircraft's operational capability is seriously reduced.

## HANDLING QUALITIES

### Control Positions in Trimmed Forward Flight

25. The trimmed control positions were determined in forward flight in configuration A (basic aircraft), configuration B (basic aircraft with IRCM exhaust stacks), and configuration E (LR-painted fuselage, modified rotor, and IRCM exhaust stacks). The tests were conducted with the helicopter trimmed in zero sideslip steady-heading level flight in 10-knot increments from 25 knots calibrated airspeed (KCAS) to 105 KCAS. As depicted in figure 27, appendix G, there was essentially no difference in control positions (less than 1 inch) between the configurations tested. All control margins were adequate and the requirements of MIL-H-8501A concerned with control margin were met. Within the scope of this test, the LR paint on the fuselage and rotor blades and/or the addition of the IRCM exhaust stacks had no significant effect on control positions in trimmed forward flight.

### Static Longitudinal Stability

26. Static longitudinal stability of the OH-58A helicopter was evaluated by varying the airspeed in 5-knot increments using the longitudinal cyclic control while maintaining the collective fixed at the initial trim position. Control positions were recorded while the helicopter was stabilized at each incremental airspeed above and below the trim airspeed. Test results are presented in figures 28 through 31, appendix G.

27. For all configurations tested (A, B, and C), the aircraft possessed positive static longitudinal stability as evidenced by the forward longitudinal control displacement for increased airspeed and aft longitudinal control displacement for decreased airspeed from trim. In configuration C (LR-painted fuselage and rotor blades and IRCM exhaust stacks) more forward longitudinal control was required at trim than in the other configurations, with no discernible change in control position gradient; however, the maximum difference in control positions between configurations was less than 1 inch and was not considered significant.

28. There was less than 1 inch of lateral cyclic and tail rotor control (pedal) variation between configurations (figs. 28 through 31, app G). Variations in the lateral cyclic movement and pedal travel with changes in airspeed between configurations were insignificant. The requirements of MIL-H-8501A for static longitudinal stability were met for all conditions tested.

### Static Lateral-Directional Stability

29. Static lateral-directional stability was evaluated under the flight conditions detailed in table 6. The tests were conducted by establishing a trim airspeed with zero sideslip and incrementally varying the sideslip angle while maintaining constant airspeed, collective control, and steady heading. Control positions were recorded at each stabilized steady-heading sideslip angle. The results are presented in figures 32 through 34, appendix G. As shown, there is no significant difference between configurations.

30. Static directional stability was positive for all airspeeds and configurations, as indicated by the directional control position versus sideslip gradient. However, the shallow gradient near trim (zero sideslip) at low airspeeds causes poor directional stability. The poor directional stability permits a residual directional oscillation noted during flight at low airspeeds. This characteristic is common to the OH-58A helicopter and is not attributable to the LR paint or IRCM exhaust system.

31. Side force, as indicated by the bank angle versus sideslip gradients (figs. 32 through 34, app G), increased with an increase in airspeed. At low airspeed, the side forces were light and did not provide the pilot adequate cues to small sideslip excursions. At higher airspeeds, side force was sufficient to prevent the pilot from inadvertently flying in a sideslip. Although side-force cues were insufficient at low airspeeds, the allowable sideslip limit would not normally be exceeded. Within the scope of this test, side-force characteristics are satisfactory.

Table 6. Static Lateral-Directional Stability.<sup>1</sup>

Configuration	Density Altitude (ft)	Outside Air Temperature (°C)	Gross Weight (lb)	Center-of-Gravity Location (in.)	Trim Calibrated Airspeed (kt)
A	9060	18.5	2840	107.8 (fwd)	85
A	8400	22.0	2780	107.6 (fwd)	40
B	7060	21.5	2780	107.6 (fwd)	86
C	7740	15.0	2820	107.1 (fwd)	42
C	6660	15.5	2860	107.2 (fwd)	80

<sup>1</sup>Rotor speed: 354 rpm.  
Doors-on, unarmed configuration.



32. Effective dihedral was weak in all configurations; however, it was not objectionable at the airspeeds and sideslips tested. Longitudinal control displacement changed only slightly throughout the sideslip range tested. There was no significant change between configurations.

### Maneuvering Stability

33. The maneuvering stability characteristics of the OH-58A helicopter were tested in configuration A (basic aircraft), configuration B (IRCM exhaust stacks), and configuration C (LR paint and IRCM exhaust stacks). The symmetrical pull-up was the test technique used during this test. The helicopter was trimmed in level flight at the desired airspeed, after which a cyclic pull-up to a slightly higher altitude was initiated. A dive was then established and the helicopter was accelerated to slightly higher than trim airspeed. A symmetrical pull-up was executed so as to pass through the required trim airspeed, altitude, a nose-level pitch attitude, and the desired load factor simultaneously. Longitudinal stick force, stick position, and normal acceleration were recorded at each test point; the results are presented in figure 35, appendix G, and summarized in figure C.

34. The boost-on stick force gradient relative to the acceleration (FS/g) was positive for all configurations tested, i.e., a greater aft cyclic force was required for an increased load factor. Considerably less stick force and travel were required with configuration C for the same amount of g force than with the basic aircraft. Although stick-force-per-g gradients were light in configuration C, it is not likely that the helicopter operational flight envelope will be exceeded during normal operation because of the cues provided by the 2-per-rotor-revolution (2/rev) vertical vibration (impending blade stall) near 2g and the large pitch attitudes required. The OH-58A maneuvering stability characteristics are satisfactory for all configurations tested.

### Dynamic Stability

35. Longitudinal, lateral, and directional dynamic stability characteristics of the OH-58A helicopter were tested under the conditions listed in table 7. Representative results are presented as time histories in figures 36 through 51, appendix G.

36. The longitudinal long-period stability characteristics were evaluated by stabilizing the helicopter at trim airspeed and then increasing or decreasing the airspeed in desired increments, using only the cyclic control. The cyclic was then returned to the trim position and the helicopter response was recorded on magnetic tape. The long-period oscillatory mode was convergent for all level flight conditions (figs. 36 through 38, app G) except at low airspeeds (40 KTAS trim), where a neutral long-period oscillation was noted (figs. 39 and 40). The period of oscillation was approximately 23 seconds. The oscillation was easily excited, resulting in moderate pilot compensation required to maintain 40 knots indicated airspeed (KIAS) (HQRS 4). The easily excitable longitudinal long-term oscillation at low airspeeds (approximately 40 KIAS) occurring in all configurations tested

FIGURE C  
MANEUVERING STABILITY  
JOM-58A  
S/N USA 68-16706

NOTE: 1 Configuration A denotes std paint, std stacks, std rotor  
2 Configuration B denotes std paint, LR stacks, std rotor  
3 Configuration C denotes LR paint, LR stacks, LR rotor

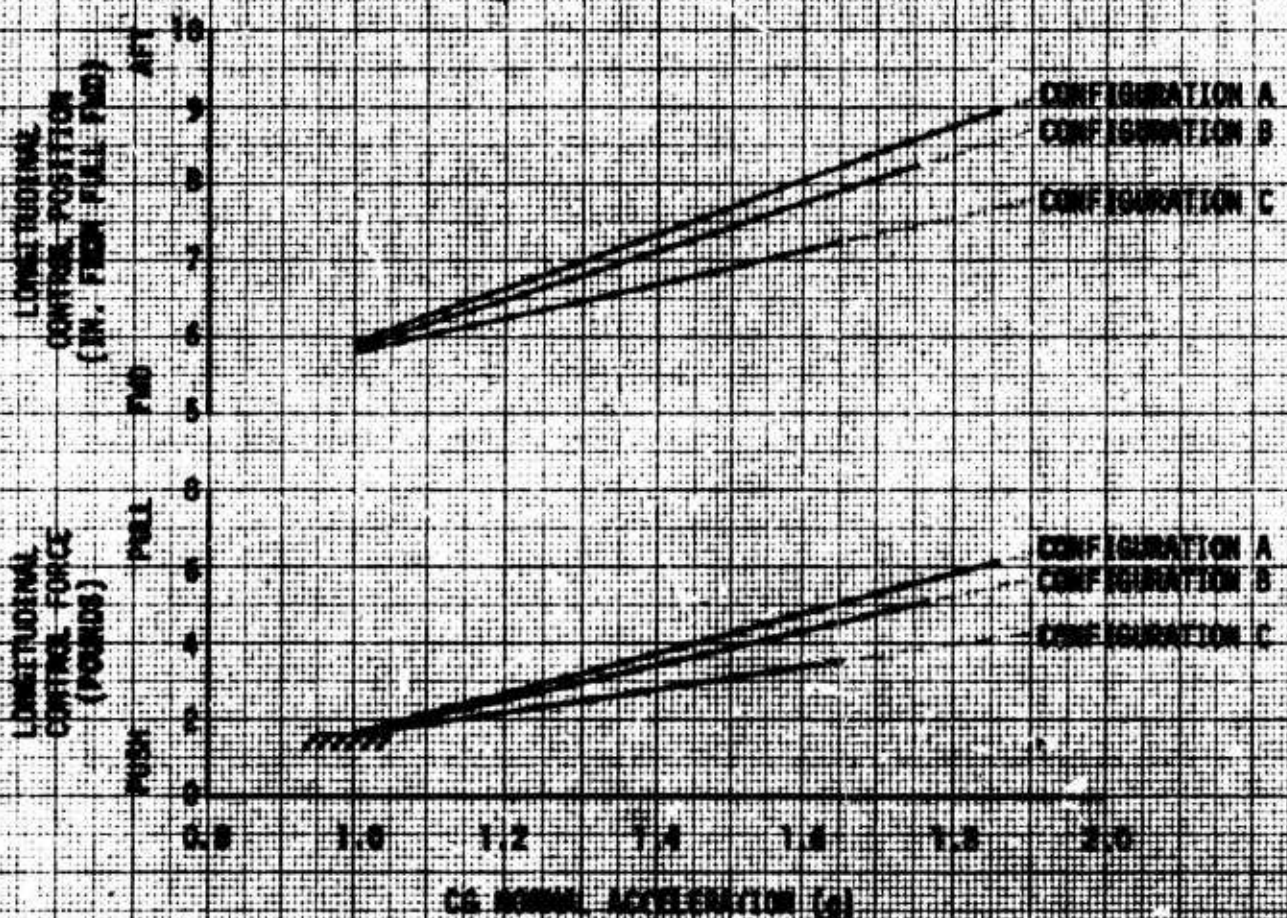


Table 7. Dynamic Stability.<sup>1</sup>

Test Description	Density Altitude (ft)	Outside Air Temperature (°C)	Gross Weight (lb)	Center-of-Gravity Location (in.)	Trim Calibrated Airspeed (kt)	Configuration
Longitudinal long-term response	7680	24.0	2800	107.6 (fwd)	52	A
	7480	23.5	2780	107.6 (fwd)	88	A
	5860	17.0	2920	107.7 (fwd)	41	C
	7780	14.0	2720	107.5 (fwd)	42	C
Longitudinal pulse	7200	14.0	2780	107.5 (fwd)	88	C
Release from steady sideslip	8340	42.5	2740	107.6 (fwd)	42	A
	8320	42.0	2720	107.6 (fwd)	85	A
	6580	20.5	2760	107.6 (fwd)	83	B
	8100	14.0	2800	107.6 (fwd)	42	C
	8340	14.0	2800	107.6 (fwd)	85	C
	8400	22.0	2720	107.6 (fwd)	87	A
	6260	16.0	2840	107.6 (fwd)	87	C
	7220	14.0	2800	107.6 (fwd)	87	C
	6940	14.5	2800	107.6 (fwd)	89	C
	6280	15.0	2820	107.6 (fwd)	91	C

<sup>1</sup>Rotor speed for all tests: 354 rpm.  
Aircraft flown with doors on in the unarmed configuration.

is a shortcoming of the basic OH-58A which is unaffected by the addition of LR paint or IRCM exhaust stacks. Within the scope of this test, there was no discernible difference between configurations tested.

37. The longitudinal short-period characteristics of the OH-58A helicopter were tested by applying cyclic pulse inputs to excite the aircraft short-period mode. As indicated in figure 41, appendix G, the short-period mode was essentially deadbeat and representative of all configurations tested.

38. The dynamic lateral-directional stability characteristics of the OH-58A helicopter were tested in level flight under the conditions shown in table 7. Gust response tests were conducted by applying 1.5-inch lateral and directional control inputs (doublets) to the controls independently in both directions for 0.5 second duration. Additionally, releases from steady-heading sideslip were made to excite the Dutch-roll characteristics.

39. In all configurations the OH-58A had a lightly damped lateral-directional oscillation (Dutch-roll) (figs. 42 through 46, app G) and a neutral spiral stability (figs. 47 through 51). The roll-to-yaw ratio was approximately 0.5 ( $\phi/\beta = 0.5$ ) and the period was approximately 4 seconds. The inherent lateral-directional oscillations were annoying and required a moderate amount of pilot compensation (HQRS 4) to maintain a constant heading. Although a shortcoming, the lateral-directional oscillation characteristics were not altered by the addition of IRCM exhaust stacks or LR paint.

40. Flight was made into light-to-moderate turbulence of varying degrees in configurations A and C at 80 KCAS (operator's manual recommended turbulence penetration airspeed). Considerable pilot effort (HQRS 5) was required to maintain straight and level flight as airspeed varied 10 knots, pitch attitude changed  $\pm 3$  degrees, roll  $\pm 4$  degrees, and sideslip varied  $\pm 5$  degrees. The constant attitude changes resulted in considerable pilot compensation and made flying in such conditions difficult. The amount of attitude and airspeed excursions did not perceptibly change between configurations. However, the neutral long-term response discussed in paragraph 36 will increase pilot workload should the flight be made in turbulence at low airspeeds (40 to 50 KCAS).

#### Simulated Sudden Engine Failure

41. Simulated sudden engine failure tests (throttle chops) were conducted under the conditions detailed in table 8. To simulate a sudden engine failure, the helicopter was trimmed at a given flight condition and the throttle abruptly closed to the flight-idle position. The flight controls were held fixed until either a minimum transient rotor speed of 304 rpm was reached or until 2 seconds had elapsed. The delay in moving the controls was to simulate the normal delay in pilot reaction time following an actual engine failure. The resultant pitch, roll, and yaw excursions were recorded, as were rotor speed decay rates. The results of these tests, including time history plots, are presented in figures 52 through 55, appendix G.

Table 8. Simulated Sudden Engine Failure.<sup>1</sup>

Configuration	Density Altitude (ft)	Outside Air Temperature (°C)	Gross Weight (lb)	Center-of-Gravity Location (in.)	Trim Calibrated Airspeed (kt)
A	8300	21.0	2720	107.5 (fwd)	52
A	7660	21.0	2740	107.5 (fwd)	97
C	5960	16.5	2880	107.5 (fwd)	51
C	6200	16.0	2880	107.5 (fwd)	51
C	5660	17.0	2860	107.5 (fwd)	52
C	6020	16.0	2860	107.5 (fwd)	96
C	5940	16.0	2860	107.5 (fwd)	96

<sup>1</sup>Rotor speed: 354 rpm.  
Door-on, unarmed configuration.

42. Aircraft response varied between configurations. The most notable difference was rotor speed decay rate between configuration A (basic aircraft) and configuration C (LR paint and IRCM exhaust stacks). For configuration A the rate of decay was 16.0 rpm/sec and for configuration C 29.0 rpm/sec. In configuration A after lowering the collective, rotor speed decayed an additional 18 rpm (fig. 52, app G) and in configuration C an additional 20 rpm (fig. 54).

43. Should a sudden engine failure occur in configuration C with a 2-second delay as specified in MIL-H-8501A, the rotor speed can be expected to reach a minimum of 282 rpm, which is 22 rpm below the minimum allowable as specified in reference 12, appendix A. A rotor speed decay of such magnitude (72 rpm) may prevent the pilot from performing a safe autorotational entry and landing while observing the existing operator's manual height-velocity diagram. If the OH-58A rotor blades are to be painted with LR paint, additional autorotational testing should be conducted, to include height-velocity testing. The autorotational rotor speed decay rate and the minimum transient rotor speed encountered prevent the OH-58A from meeting the requirements of paragraph 3.5.5 of MIL-H-8501A. Although the standard OH-58A has reportedly failed to meet the specification requirements in previous tests, nonconformance with the specification is increased with the addition of LR paint. The excessive rotor speed decay rate with LR-painted rotor blades is a deficiency.

## **SUBSYSTEM TESTS**

### **Engine Characteristics**

44. A limited evaluation of the effects of IRCM exhaust stacks on engine performance was conducted and the results are shown in figure 56, appendix G. A single static probe was located 2 inches downstream from the marmon clamp on the inside of the left exhaust stack, as shown in photo C, appendix B. The high-pressure readings probably resulted from interference caused by the close proximity of the tap to the clamp. The pressure measured is approximately equal to the dynamic pressure ( $Pv^2/2g$ ) and is probably not representative of the average static pressure at other circumferential locations. Meaningful data were difficult to obtain because of the close proximity of the tap to the flange clamp joint and the tangential velocities induced by the gas turbine. Further testing should be performed on a more fully instrumented exhaust installation to ascertain the true exhaust static pressure and its effect on engine performance.

45. Referred engine parameters were used to compare the test engine with the standard BHC exhaust extension against the same engine with the Hughes IRCM exhaust stacks installed. Data on referred gas producer speed, shp, turbine outlet temperature, and fuel flow are presented in figures 57 through 59, appendix G.



46. The referred shp versus referred gas producer speed shows that the gas producer speed of the engine equipped with the IRCM exhaust extension was increased over that of the standard ducts. At a gas producer speed equal to 100 percent of rpm, the engine equipped with IRCM exhaust stacks produced 3 shp less than the engine equipped with the standard ducts (fig. 57, app G). Referred shp versus referred turbine outlet temperature shows that the engine runs 6.5°C hotter (at the same power setting) when equipped with the IRCM exhaust stacks than with the standard ducts. At the turbine outlet temperature for normal rated power of 693°C, the engine when equipped with IRCM exhaust stacks produced 6.5 shp less (6.5 pounds more of fuel per hour at the same power setting) than the engine equipped with standard ducts (figs. 58 and 59).

47. The Hughes IRCM exhaust stacks have degraded the engine performance by approximately 2 percent. The engine model specification power and fuel flow (ref 11, app A) for the T63-A-700 engine were modified to reflect the installation of the IRCM exhaust stacks as presented in figures 60 through 62, appendix G. These corrected powers and fuel flows were used to calculate the hover performance summaries and the level flight range data presented in this report.

#### Engine Compartment Temperature Survey

48. A limited engine compartment temperature survey was performed with the Hughes IRCM exhaust stacks installed. Temperatures were recorded at locations specified in Allison Installation Design Manual (ref 11, app A). The temperature probes were located at the following locations: compressor section, top engine mount pad surface, ignition harness, thermocouple harness, and oil heat exchanger. Temperature data for all conditions tested are presented in figure 63, appendix G. When the component/fluid temperature data are corrected to the Army's design requirement maximum ambient temperature of 125°F (52°C), no overtemperature condition is indicated, as is shown in figure 65, appendix G. The temperature data were corrected by the following equation:

$$T = T_{\text{measured}} \frac{52 + 273}{T_{\text{ambient}} + 273}$$

(all temperatures are in degrees Centigrade)

The effect of altitude-temperature variation was not determined. Within the scope of this evaluation, the engine compartment and oil heat exchanger cooling were satisfactory.

# **CONCLUSIONS**

## **GENERAL**

49. The following general conclusions were reached upon completion of testing:

a. Painting the rotor blades with LR paint degraded the hover and level flight performance to an extent that the aircraft's operational capability is seriously reduced.

b. The configuration of LR-painted fuselage, standard rotor, and IRCM exhaust stacks resulted in a significant level flight performance reduction.

c. One deficiency and two shortcomings were identified during the evaluation.

## **DEFICIENCY AND SHORTCOMINGS**

50. The following deficiency was identified: the excessive rotor speed decay rate with LR paint on the rotor blades (para 43).

51. The following shortcomings were identified:

a. Easily excited longitudinal long-term oscillation at low airspeeds (approximately 40 KIAS) unaffected by the addition of LR paint or IRCM exhaust stacks (para 36).

b. Inherent lateral-directional oscillations requiring moderate pilot compensation to maintain a constant heading were unaffected by the addition of LR paint or IRCM exhaust stacks (para 39).

## **SPECIFICATION COMPLIANCE**

52. Within the scope of this test, the OH-58A helicopter failed to meet the requirements of paragraph 3.3.5 of MIL-H-8501A, in that the 2-second time delay following a simulated sudden engine failure was not achievable in a maximum power climb at 51 KCAS for all configurations tested (para 43).



## **RECOMMENDATIONS**

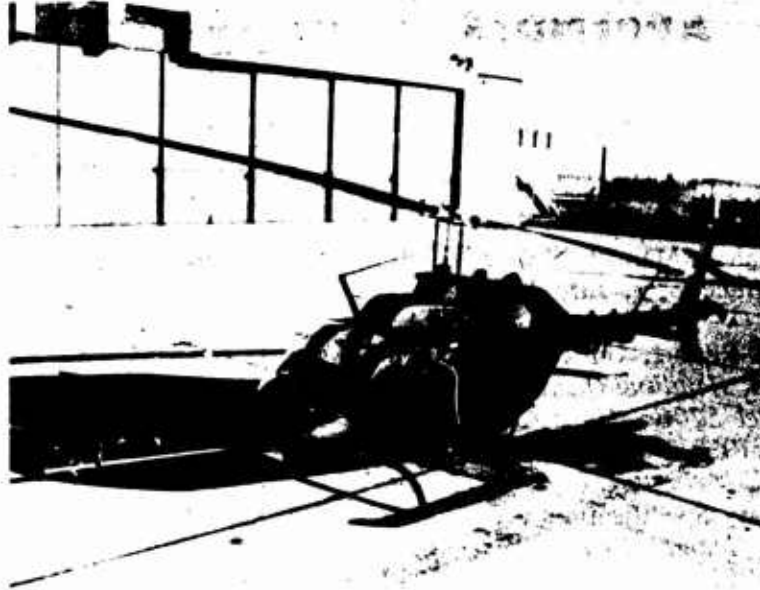
53. If the OH-58A rotor blades are to be painted with LR paint, additional autorotational evaluations should be conducted, to include height-velocity testing (para 43).

54. Further testing should be performed on a more fully instrumented exhaust installation to ascertain the true exhaust static pressure and its effect on engine performance (para 44).

## APPENDIX A. REFERENCES

1. Letter, AVSCOM, AMSAV-EFT, 5 November 1974, subject: OH-58A Suppressor DT II Army Preliminary Evaluation, as amended by telephone message to include LR paint.
2. Technical Manual, TM 55-1520-288-10, *Operator's Manual, Army Model OH-58A Helicopter*, July 1969.
3. Final Report, United States Army Aviation Systems Test Activity, Project No. 72-20, *Handling Qualities Evaluation of the OH-58A Helicopter Incorporating the Model 570B Three-Axis Stability and Control Augmentation System*, February 1973.
4. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities; General Requirements For*, 7 September 1961, with Amendment 1, 3 April 1962.
5. Final Report, United States Army Aviation Systems Test Activity, Project No. 68-30, *Airworthiness and Flight Characteristics Evaluation, Engineering Flight Test of a Production OH-58A Helicopter Unarmed and Armed with XM27E1 Weapon Subsystem, Performance*, September 1970.
6. Final Report, United States Army Aviation Systems Test Activity, Project No. 72-01, *Instrument Flight Evaluation, OH-58A Helicopter*, September 1972.
7. Message, AVSCOM, AMSAV-EQI, 3 June 1975, subject: Safety of Flight Release, Project No. 75-11.
8. Flight Test Manual, Naval Air Test Center, FTM No. 101, *Stability and Control*, 10 June 1968.
9. Engineering Design Handbook, Army Materiel Command Pamphlet No. 706-204, *Helicopter Performance Testing*, August 1974.
10. Letter, AVSCOM, AMCPM-ASE-TM, 3 June 1975, subject: IR Low Reflective Paint for OH-58 Helicopter.
11. Model Specification, Allison Division of General Motors Corporation, No. 803, "Allison Model T63-A-700," 19 July 1968.
12. Detail Specification, Bell Helicopter Company, No. 206-947-031, "Model 206A (Mod)," 16 October 1967.
13. Flight Test Manual, USAF Aerospace Research Pilot School, FTC-TIH-68-1002, *Stability and Control*, September 1968.

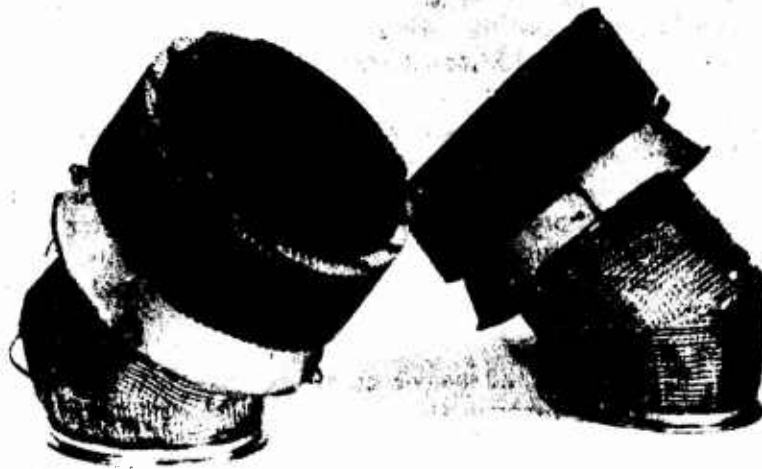
## **APPENDIX B. PHOTOGRAPHS**



**Photo A. Aircraft in Configuration A.**



**Photo B. Aircraft with IRCM Exhaust Stacks Installed.**



**Photo C. Hughes IRCM Exhaust Stacks.**

## **APPENDIX C. DESCRIPTION**

### **INFRARED COUNTERMEASURE EXHAUST STACKS**

1. The prototype exhaust stacks are interchangeable with the standard exhaust stacks (P/N 206-061-300) on the aircraft. No modification to the airframe is required to receive the IRCM exhaust stacks.

2. The stacks are constructed in two sections: the nozzle portion is constructed of stainless steel and insulated with 0.25 inch of "Low Q" insulation, and the shielding section is made of aluminum with crimped fins brazed to the exterior. The shielding section is approximately 2 inches larger in diameter than the nozzle section and is mounted on the nozzle by four heat-resistant bolts. The two sections are concentrically located relative to each other and the nozzle protrudes about 0.75 inch into the shield.

3. This configuration uses the nozzle as an exhaust pipe and as an aspirating pump to draw cooling air through the annular opening between the two sections. The cooling air acts as a boundary layer, helping to protect the shield section from the hot exhaust gases. The shield is also cooled by the flow of ambient air over the cooling fins. The nozzle has a 40-degree bend as viewed from the front of the aircraft, allowing a transition from the axis of the engine outlet to a vertical axis for countermeasure purposes.

4. The system was designed to the following parameters:

a. Weight: 3.85 pounds per duct, which is an increase of 2.58 pounds over the standard exhaust stack.

b. Temperature: The average temperatures of exposed metal surfaces are not to exceed 300°F at maximum power, 95°F ambient, hovering out of ground effect.

c. Structure: The system is a highly damped structure which will not be subject to thermal shock under any mode of aircraft operation.

d. Engine compartment temperatures: Installation of the improved exhaust stacks shall not increase engine compartment temperatures from that of the standard configuration.

### **LOW REFLECTIVE PAINT**

5. The paint supplied for these tests is a low reflective olive-drab acrylic lacquer (FSN 8010-083-6588) specified by MIL-L-46159. It is specially formulated to reduce IR solar reflections in the spectral band pass of all currently identified IR-seeking missiles. It is also designed to have a low visual gloss to aid in visual contrast reduction. This paint is a substitute for standard Army olive-drab paint and can be applied directly over standard paint. The LR paint added 19 pounds to the gross weight of the aircraft.

## APPENDIX D. GENERAL TEST CONDITIONS

Configuration	Test	Average Gross Weight (lb)	Average Density Altitude (ft)	Average Outside Air Temperature (°C)	Average True Airspeed (kt)
A	Hover performance	2750 to 3250	4040	24.5	Zero
C		2315 to 3500	Sea level to 4150	15.5 to 27.0	
D		2450 to 3010	340	15.5	
E		2530 to 3040	2860	16.0	
F		2450 to 3110	2820	15.5	
G		2690 to 3030	2820	14.5	
A	Level flight performance	2820	7880	17.0	35 to 113
B		2820	7240	24.0	35 to 105
C		2780	2880	18.0	35 to 102
E		2460	4420	19.5	34 to 115
F		2650 to 2880	3320 to 10,940	10.5 to 24.0	23 to 115
A	Control positions in trimmed forward flight	2820	7880	17.0	35 to 113
B		2820	7240	24.0	35 to 105
C		2460	4420	19.5	23 to 115
A	Static longitudinal stability	2860 to 2940	6600 to 7520	23.0	86 and 52
B		2900 to 2940	5360 to 8260	18.0 to 23.0	52 and 92
C		2920	5940 and 6340	15.0 and 26.0	53 and 82
A	Static lateral-directional stability	2780 and 2840	8400 and 9060	18.5 and 22.0	40 and 85
B		2780	2060	21.5	86
C		2870 and 2860	7740 and 6660	15.0	42 and 80
A	Maneuvering stability	2780	8960	18.5	80
B		2760	8120	22.0	90
C		2760	8220	14.5	90
A	Longitudinal dynamic response	2800 and 2780	7680 and 7480	24.0 and 23.5	52 and 88
C		2920, 2720, and 2780	5860, 7780, 7200	17.0, 14.0, and 14.0	41, 42, and 88
A	Release from steady-heading sideslip	2720 to 2740	8340 to 8400	42.5	42 to 87
B		2760	6580	22.0	83
C		2800	6260 to 8340	14.0 to 16.0	42 to 91
A	Simulated sudden engine failure	2720	7660 to 8300	21.0	52 and 91
C		2860	6000	16.0	51 and 96

## APPENDIX E. DATA ANALYSIS METHODS

1. This appendix presents the data reduction and analysis methods used to evaluate the OH-58A performance capabilities. These equations correct test-day conditions to standard-day conditions and also provide the necessary tools with which to predict helicopter performance for various atmospheric conditions. The topics discussed include level flight, hover, and specific range.

2. Level flight performance was defined by measuring the shp required to maintain level unaccelerated flight throughout the airspeed range of the helicopter. The results of each level flight test are presented as engine shp corrected to the average test conditions for each flight, tip Mach number, and specific range as a function of airspeed.

3. Helicopter performance test data were generalized through the use of nondimensional coefficients. The purpose is to accurately define performance at conditions not specifically tested. The following nondimensional coefficients were used to generalize the level flight and hover test results obtained during this flight test program.

a. Coefficient of power ( $C_P$ ):

$$C_P = \frac{\text{SHP} \times 550}{\rho A (\Omega R)^3} \quad (1)$$

b. Coefficient of thrust ( $C_T$ ):

$$C_T = \frac{W}{\rho A (\Omega R)^2} \quad (2)$$

c. Advance ratio ( $\mu$ )

$$\mu = \frac{1.6889 \times V_T}{\Omega R} \quad (3)$$

d. Advancing tip Mach number ( $M_{tip}$ ):

$$M_{tip} = \frac{1.6889 V_T + \Omega R}{a} \quad (4)$$



**Where:**

**SHP = Engine output shp**

**550 = Conversion factor (ft-lb/sec/shp)**

**$\rho$  = Air density (lb-sec<sup>2</sup>/ft<sup>4</sup>)**

**A = Main rotor disc area (ft<sup>2</sup>)**

**$\Omega$  = Main rotor angular velocity (radian/sec)**

**R = Main rotor radius (ft)**

**W = Gross weight (lb)**

**1.6889 = Conversion factor (ft/sec/kt)**

**$V_T$  = True airspeed (kt)**

**a = Speed of sound (ft/sec)**

4. Engine output shp was determined from the engine torque pressure. Torque pressure as a function of the power output of the engine was obtained from ARADMAC engine acceptance test cell data. Horsepower transmitted by a rotating shaft was determined by the following equation:

$$\text{SHP} = \frac{2\pi \times K_t \times GR \times N_r \times Q}{33,000}$$

**Where:**

**SHP = Shaft horsepower**

**$K_t$  = Conversion factor to change measured engine torque pressure (psi) to ft-lb (from engine acceptance data)**

**GR = Gear ratio of the output shaft rotational speed to the main rotor rotational speed**

**$N_r$  = Main rotor speed (rpm)**

**Q = Engine torque pressure (psi)**

**33,000 = Conversion factor (ft-lb/min/shp)**

5. Test-day level flight power was corrected to standard-day conditions by assuming that the test-day dimensionless parameters,  $C_{p_t}$ ,  $C_{T_t}$ , and  $\mu_s$  are identical to  $C_{p_t}$ ,  $C_{T_t}$ , and  $\mu_t$ , respectively. From the definition of equation 1, the following relationship can be derived:

$$SHP_s = SHP_t \times \frac{\rho_s}{\rho_t}$$

Where:

$SHP$  = Engine output shp

$\rho$  = Air density (slug/ft<sup>3</sup>)

$t$  = Test day

$s$  = Standard day

6. Specific range was calculated using the level flight performance curve and the specification installed engine fuel-flow characteristics:

$$NAMPP = \frac{V_T}{W_f}$$

Where:

$NAMPP$  = Nautical air miles per pound of fuel (naut mi/lb)

$V_T$  = True airspeed (kt)

$W_f$  = Fuel flow rate (lb/hr)

7. Changes in the equivalent flat plate area ( $f_e$ ) for various aircraft configurations were calculated by the following equation:

$$\Delta f_e = \frac{2(\Delta C_p)(A)}{\mu^3}$$

Where:

$\Delta f_e$  = Change in flat plate area (ft<sup>2</sup>)

$\Delta C_p$  = Change in coefficient of power

A = Main rotor disc area (ft<sup>2</sup>)

$\mu$  = Advance ratio

8. Hover performance was determined IGE and OGE by the free flight hover technique. Equations 1 and 2 were used to define the hover capability. A plot of  $C_p$  versus  $C_T$  was constructed for each skid height tested. Hover performance characteristics may be extracted from these curves in preparing labels or curves for flight manuals for any combination of conditions.

9. The HQRS presented as figure 1 was used to augment pilot comments relative to handling qualities.

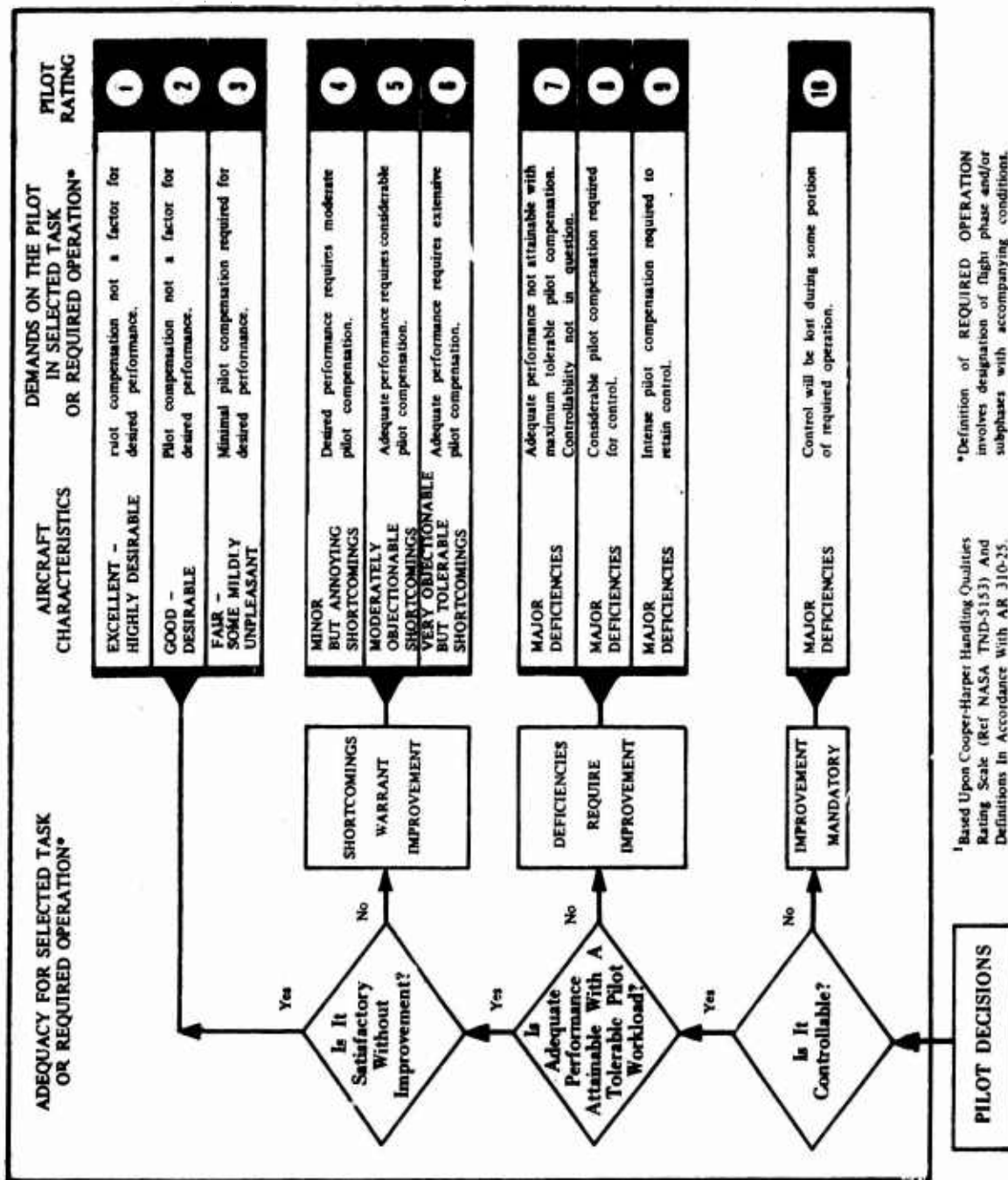


Figure 1. Handling Qualities Rating Scale.

## **APPENDIX F. INSTRUMENTATION**

Instrumentation was installed in the test helicopter by USAAEFA personnel prior to the start of the test program. One magnetic tape recorder was installed in the aft cabin compartment for the majority of all testing. The recorder was removed for the final 6 hours of testing so that light gross weights could be attained. All instrumentation was calibrated and maintained by USAAEFA personnel. The following parameters were recorded:

### **Pilot/Engineer Panel**

- Airspeed (boom)
- Altitude (boom)
- Angle of sideslip
- Rotor speed
- Center-of-gravity normal acceleration
- Free air temperature
- Total fuel used
- Turbine outlet temperature
- Engine torque
- Gas producer speed
- Fuel flow rate
- Fuel totalizer
- Engineer event marker
- Record counter
- Engine compartment temperatures
- Engine fuel nozzle pressure

### **Magnetic Tape Recorder**

Control position and force:

- Longitudinal
- Lateral
- Directional
- Collective

Attitude and rate:

- Pitch
- Roll
- Yaw

Airspeed (boom)  
Altitude (boom)  
Altitude (radar)  
Free air temperature  
Angle of sideslip  
Center-of-gravity normal acceleration  
Rotor speed  
Throttle position  
Engine torque  
Gas producer speed

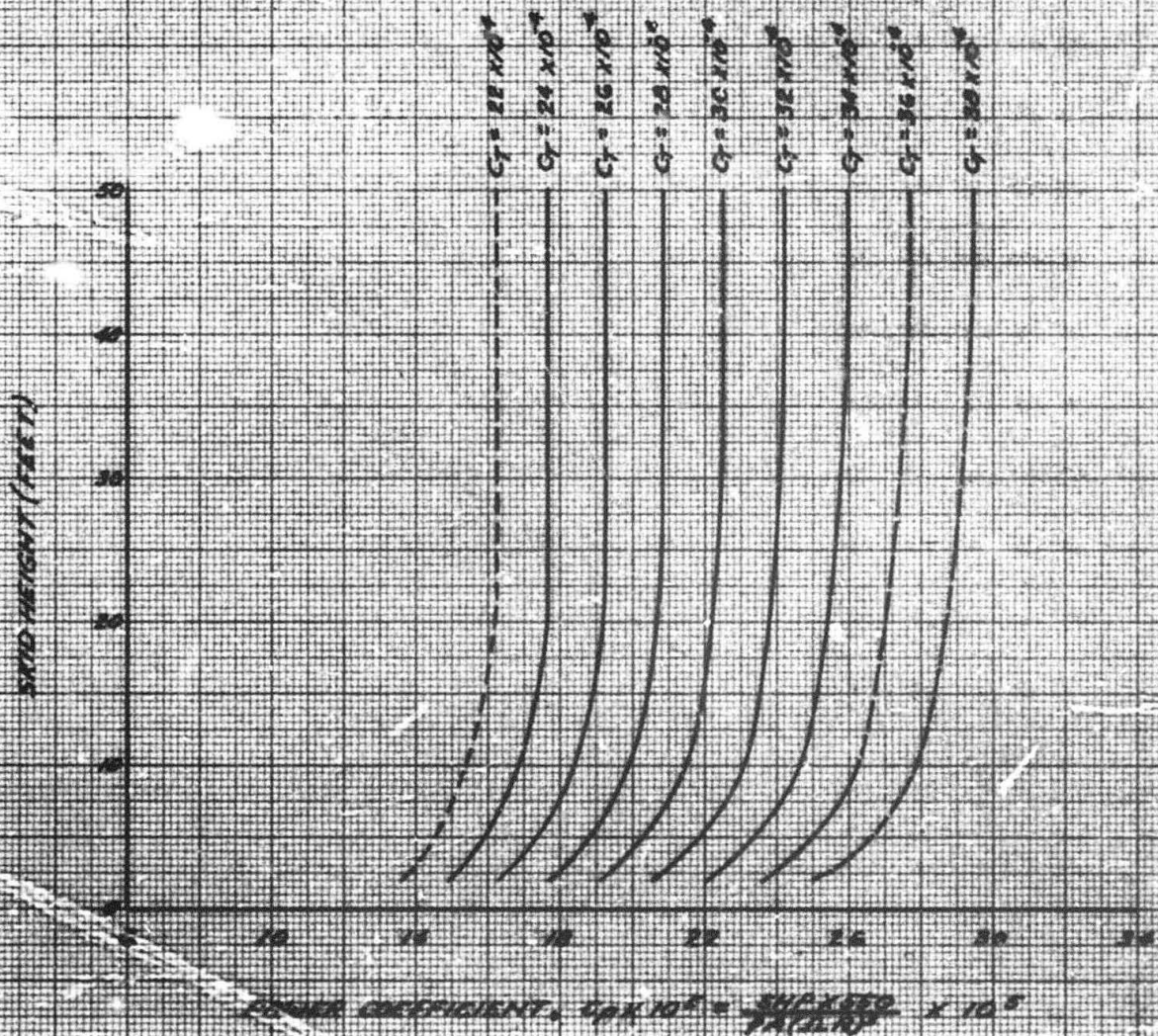
# **APPENDIX G. TEST DATA**

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Static Lateral-Directional Stability	32 through 34
Maneuvering Stability	35
Dynamic Stability	36 through 51
Simulated Sudden Engine Failure	52 through 55
<b>Subsystems Tests</b>	
Engine Characteristics	56 through 63

FIGURE 1  
 Non-Dimensional Horizontal Performance Summary  
 201-581  
 SINGLE-ENGINE  
 CONFIGURATION 2  
 FREE FLIGHT METHOD  
 WINDS LESS THAN 5 KNOTS

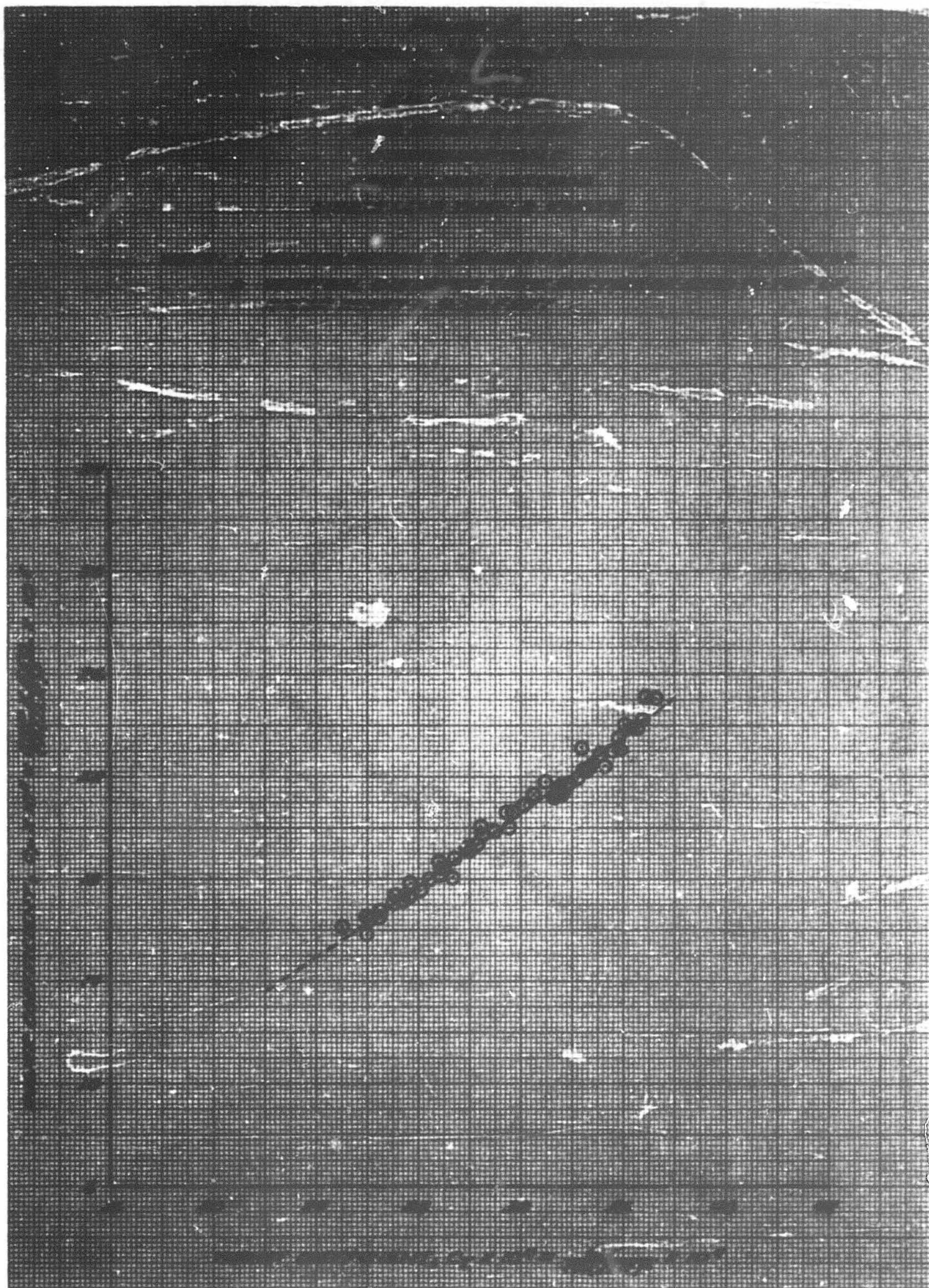
- NOTE: 1. CONFIGURATION 2 DENOTES 12.5 INCH, 6.5 INCH, 12 INCH  
 2. DERIVED FROM PLOTTED CURVES ON FIGS. 3, 4, 5, 6, 7  
 3. VERTICAL DISTANCE FROM BOTTOM OF SKID TO TOP OF  
 KITE MAST = 9.55 FEET





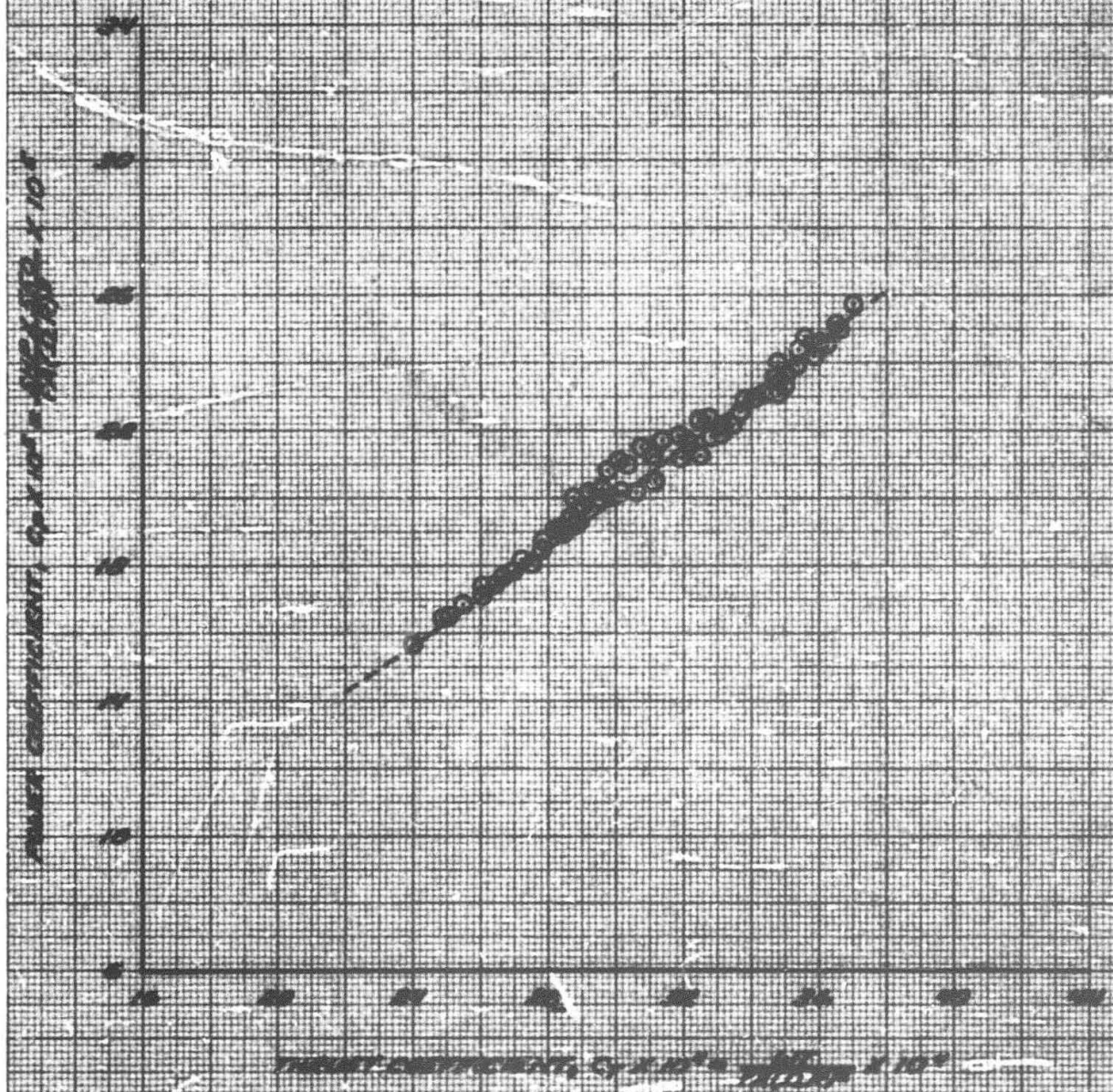
POWDER CONSUMPTION, CO X 10<sup>3</sup> - 5000000 X 10<sup>3</sup>

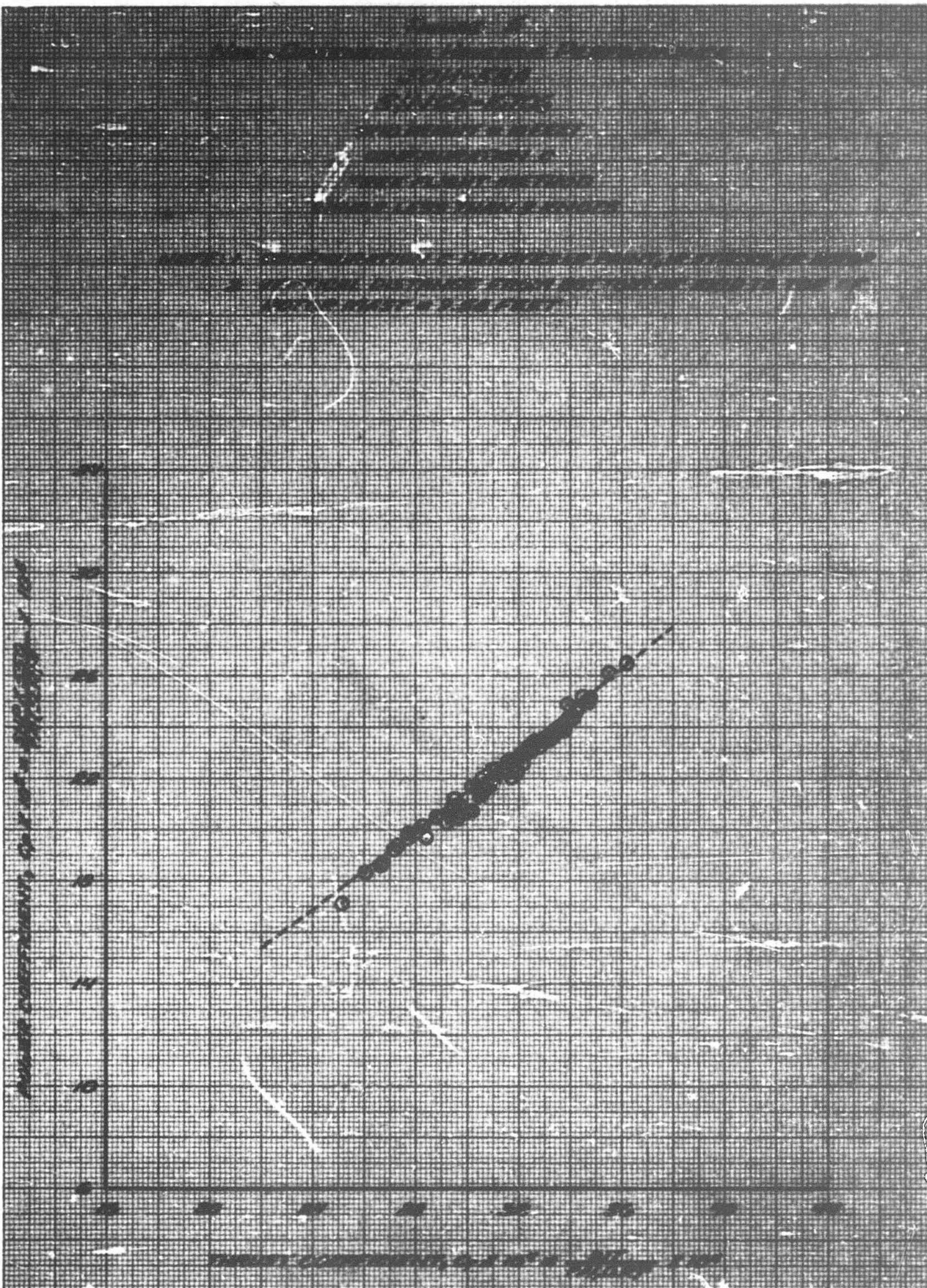
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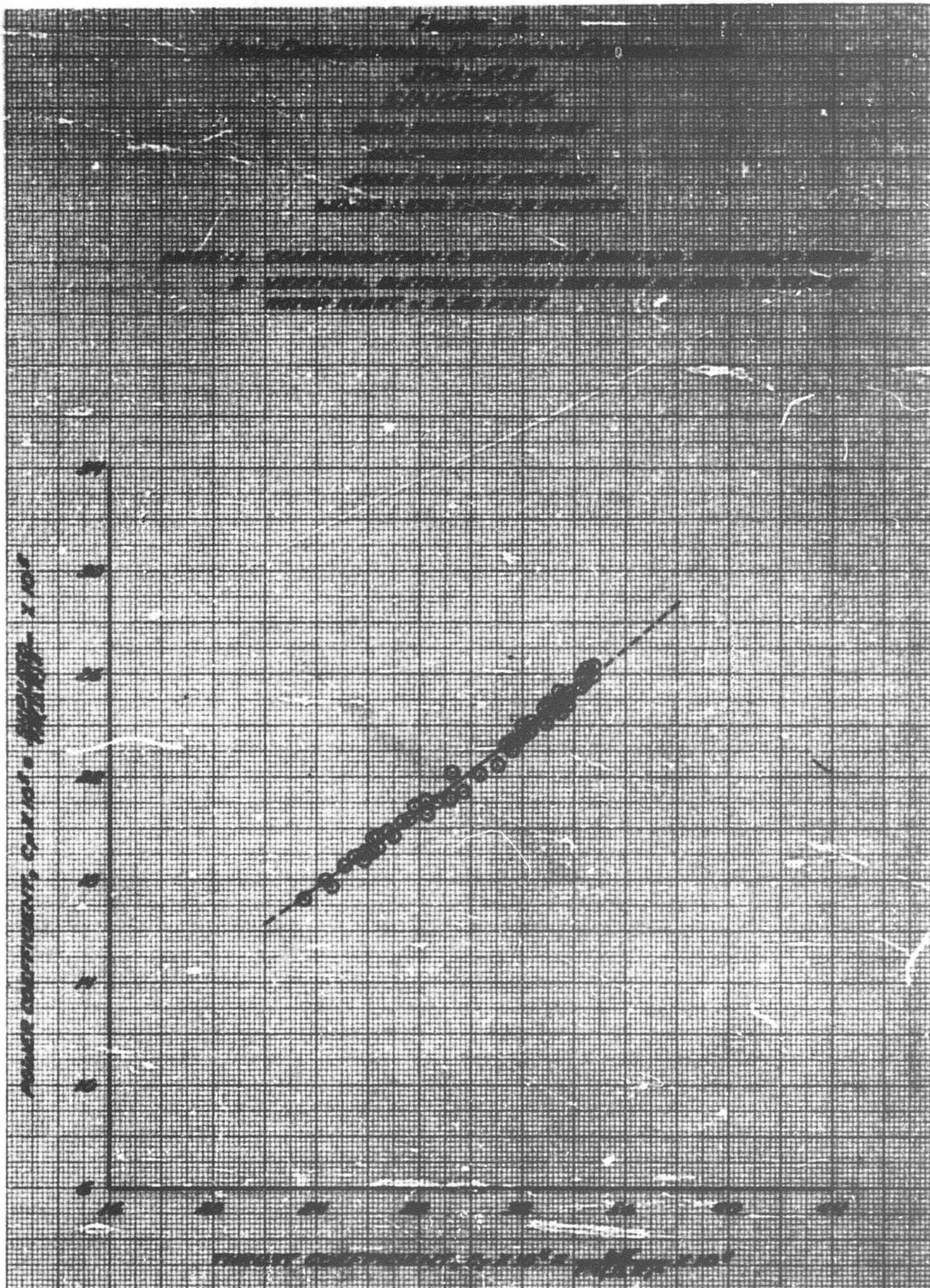


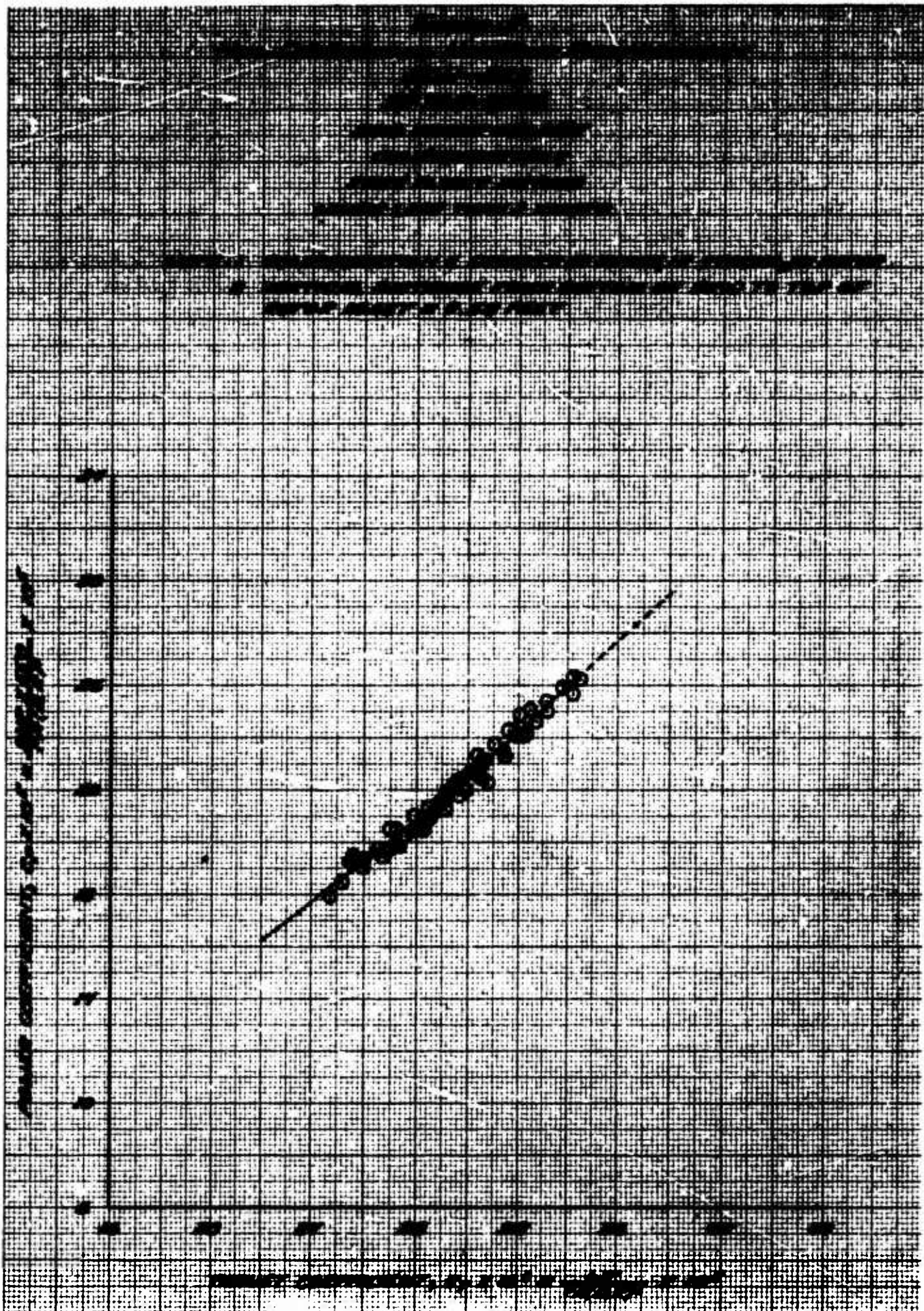
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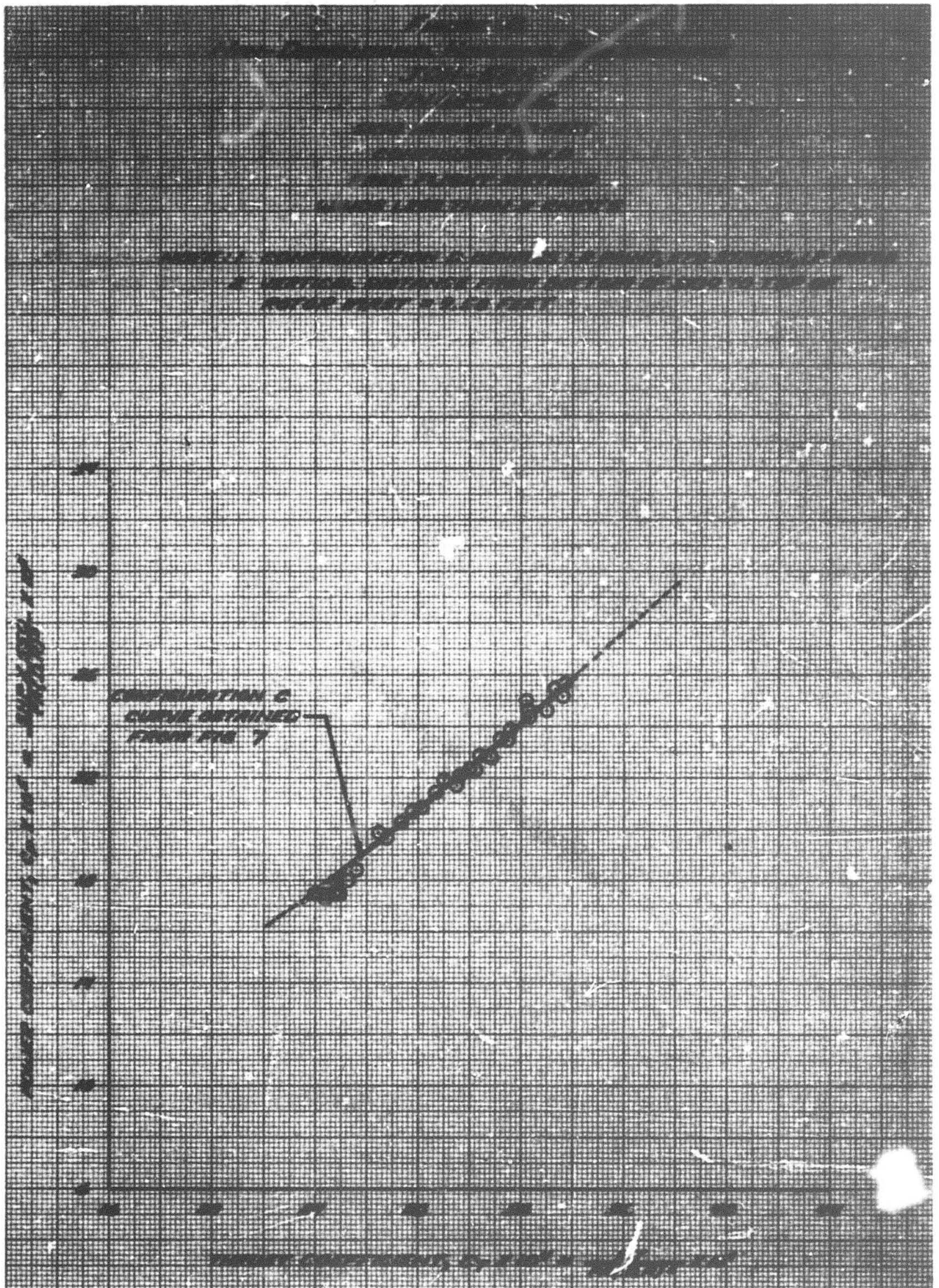


Figure 8  
 Non-Dimensional Plot of Thrust Coefficient  
 vs. Vertical Distance from Bottom of Nozzle to Top of  
 Motor Mast  
 Configuration C  
 Free Flight Method  
 Range Less Than 3 Meters

NOTE: 1. CONFIGURATION C REMAINS 1.8 FEET, IN STEADY, HANGING  
 1.8 FEET  
 2. VERTICAL DISTANCE FROM BOTTOM OF NOZZLE TO TOP OF  
 MOTOR MAST = 9.65 FEET

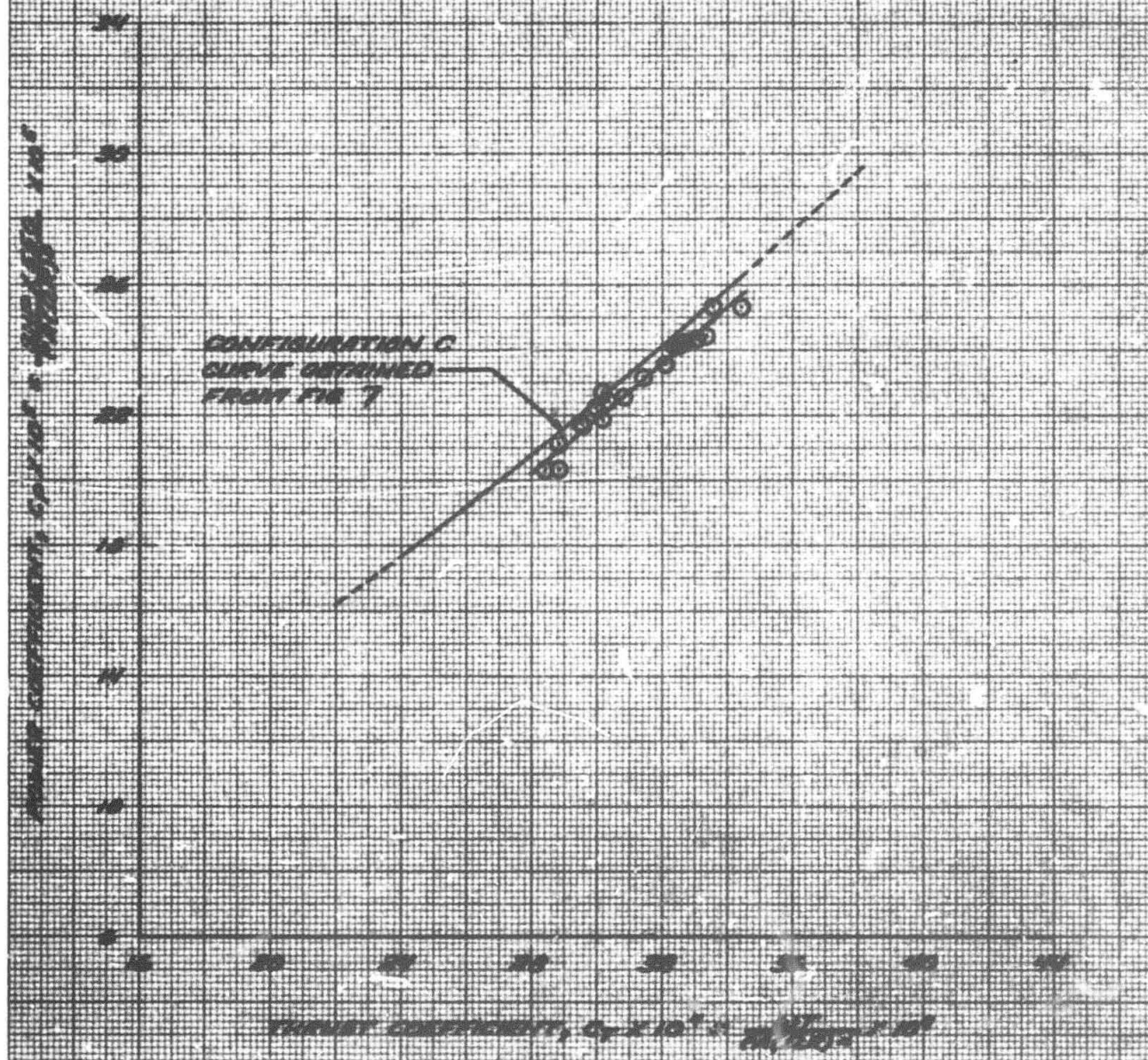
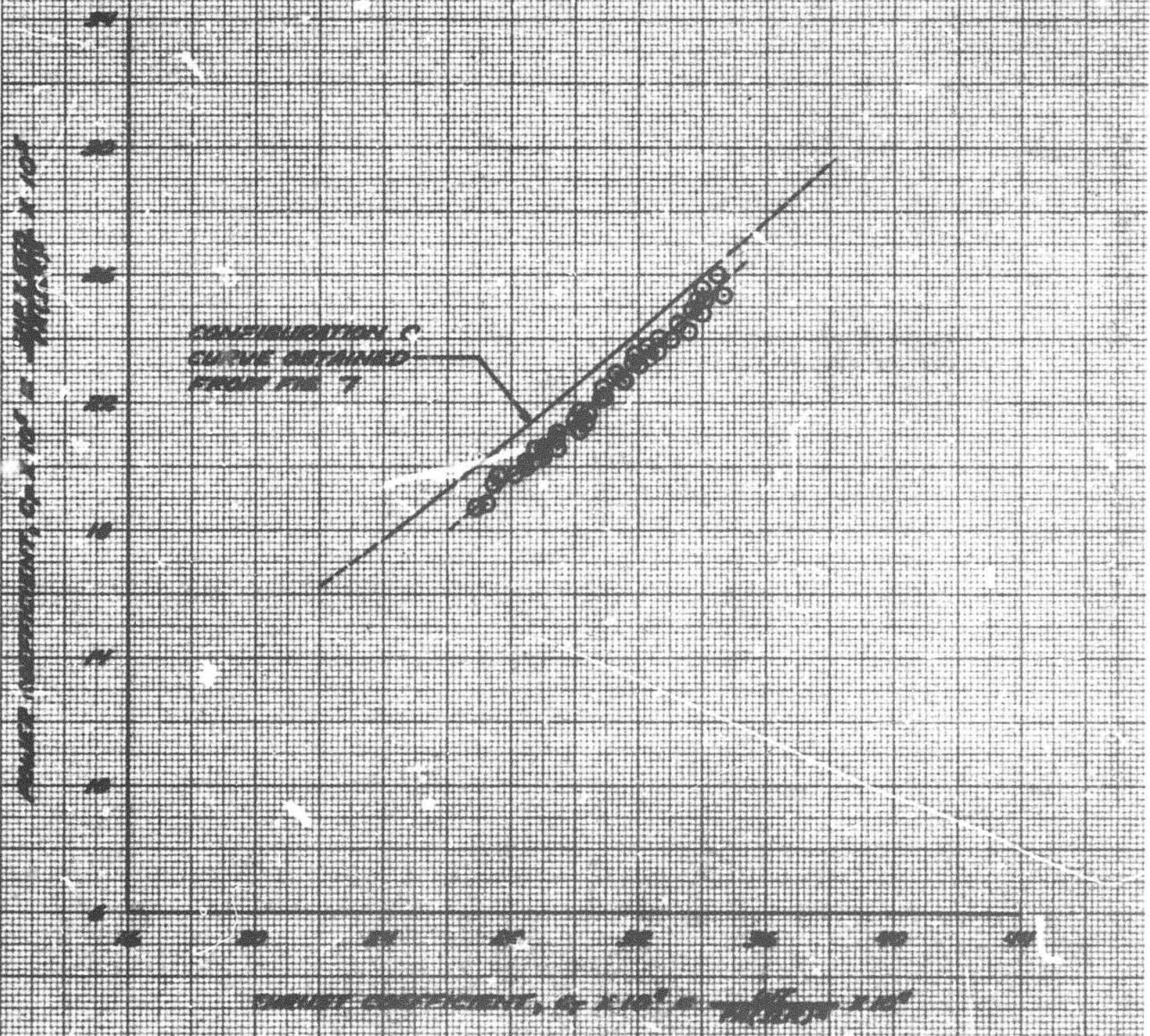


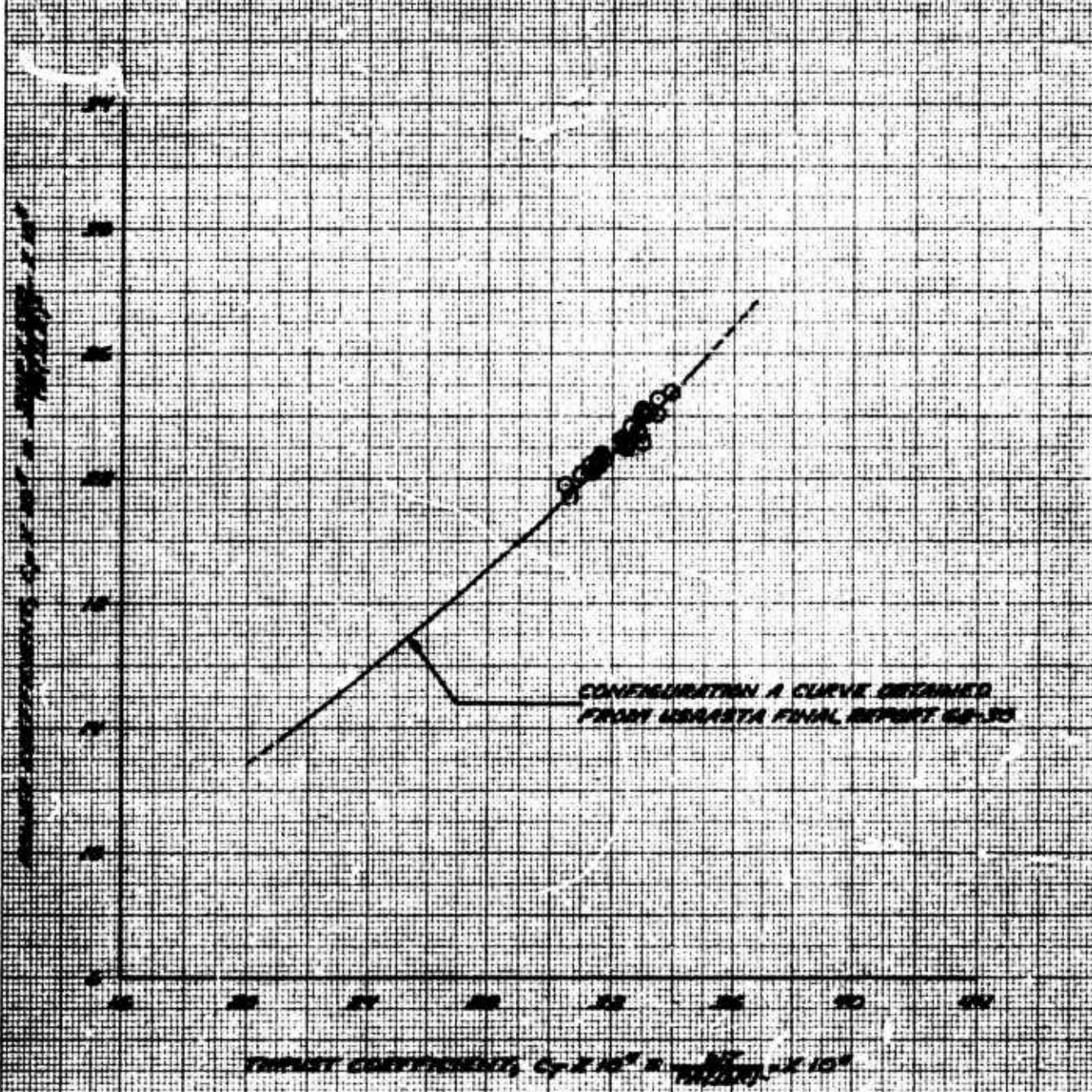


Figure 10  
 The Dependence of the Drag Coefficient  
 $C_D$  on the Angle of Attack  
 $\alpha$  for the Model  
 with a Length of 1.0 ft  
 and a Diameter of 0.5 ft  
 at a Reynolds Number of  
 $1.5 \times 10^5$

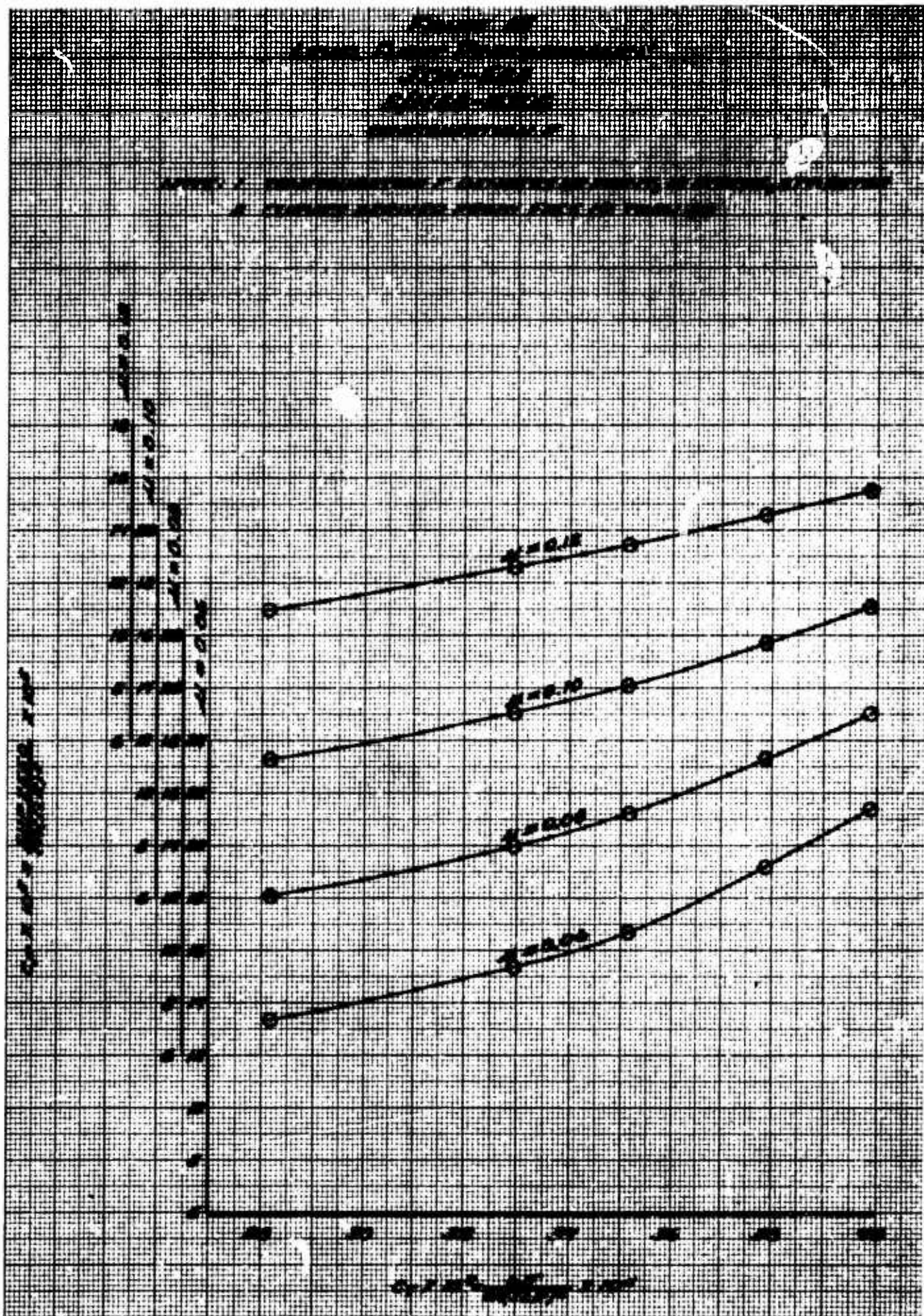
MODEL 1: CONFOURTHAL F. DRAGS 1.0 FT. IN LENGTH, STD. RADIUS  
 2. VERTICAL DISTANCE FROM BOTTOM OF SKIN TO TOP OF  
 WING RIBS = 0.50 FEET



REPORT NO. 68-35  
 1. SUMMARY OF RESULTS  
 2. INTRODUCTION  
 3. SCOPE OF STUDY  
 4. THEORY  
 5. EXPERIMENTAL PROCEDURE  
 6. RESULTS  
 7. CONCLUSIONS  
 8. REFERENCES  
 9. APPENDICES  
 10. DISTRIBUTION STATEMENTS

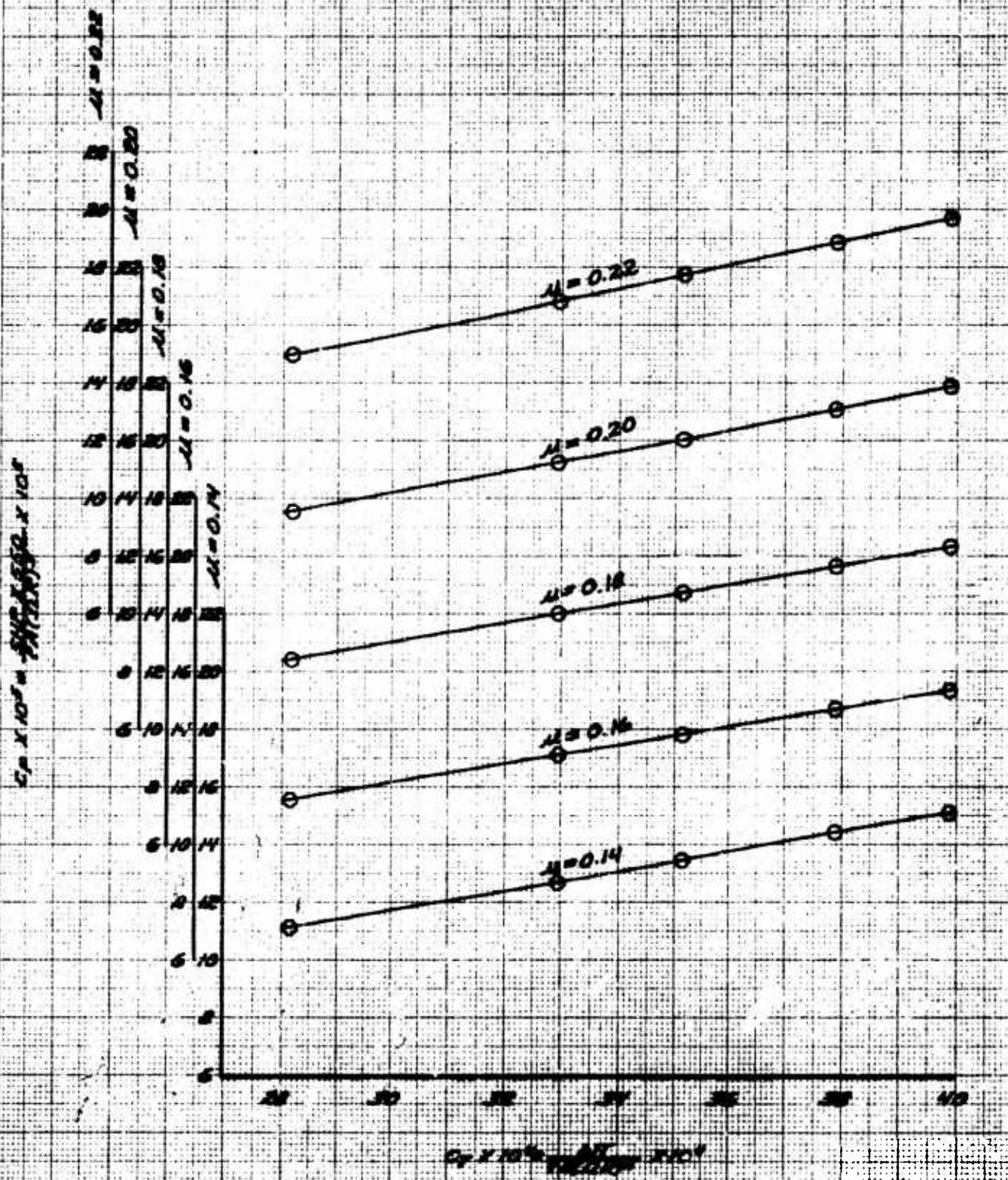






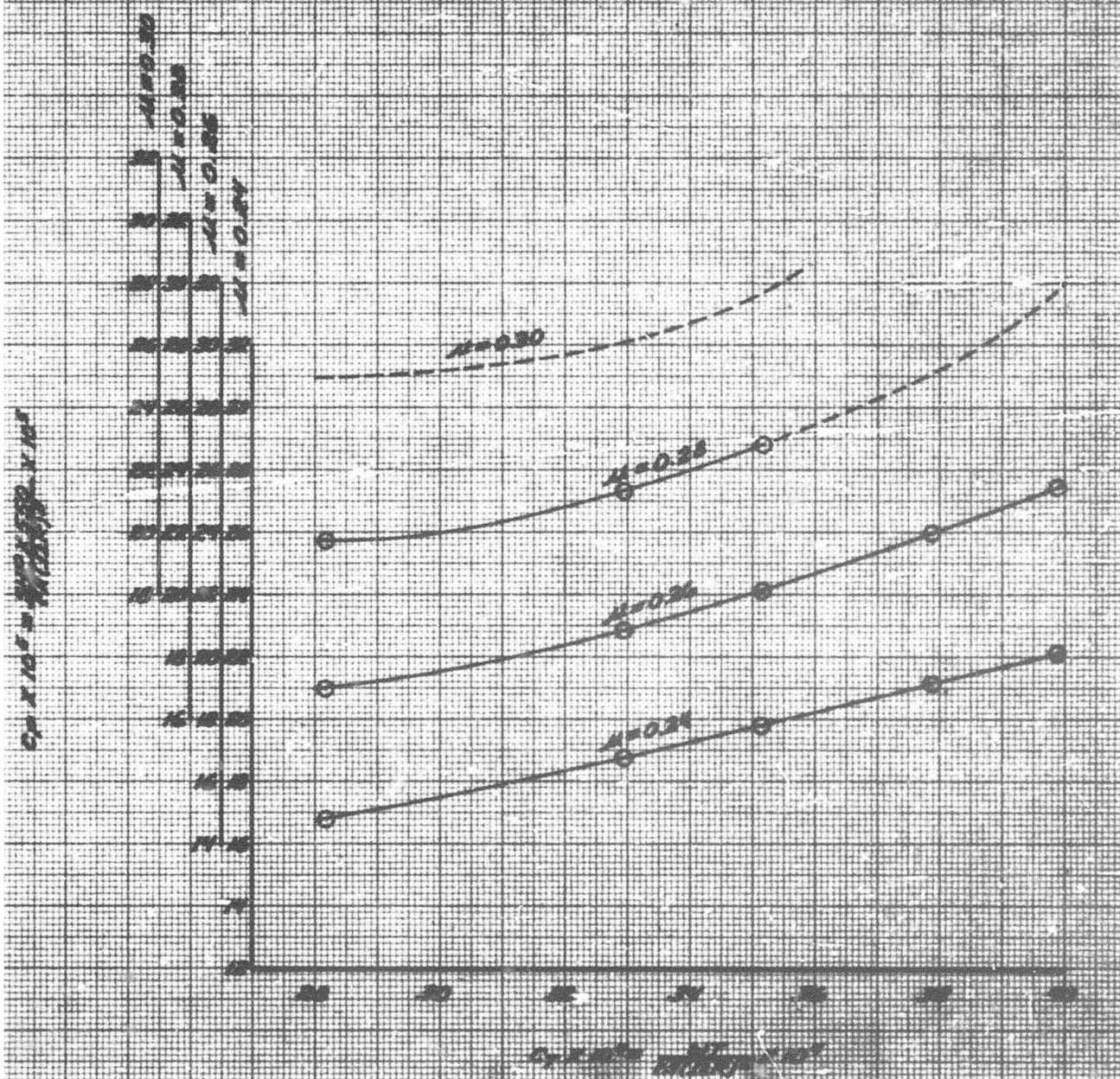
**FIGURE 13**  
**LEVEL FLIGHT PERFORMANCE**  
**JOH-58A**  
**SINGA-MTAC**  
**CONFIGURATION F**

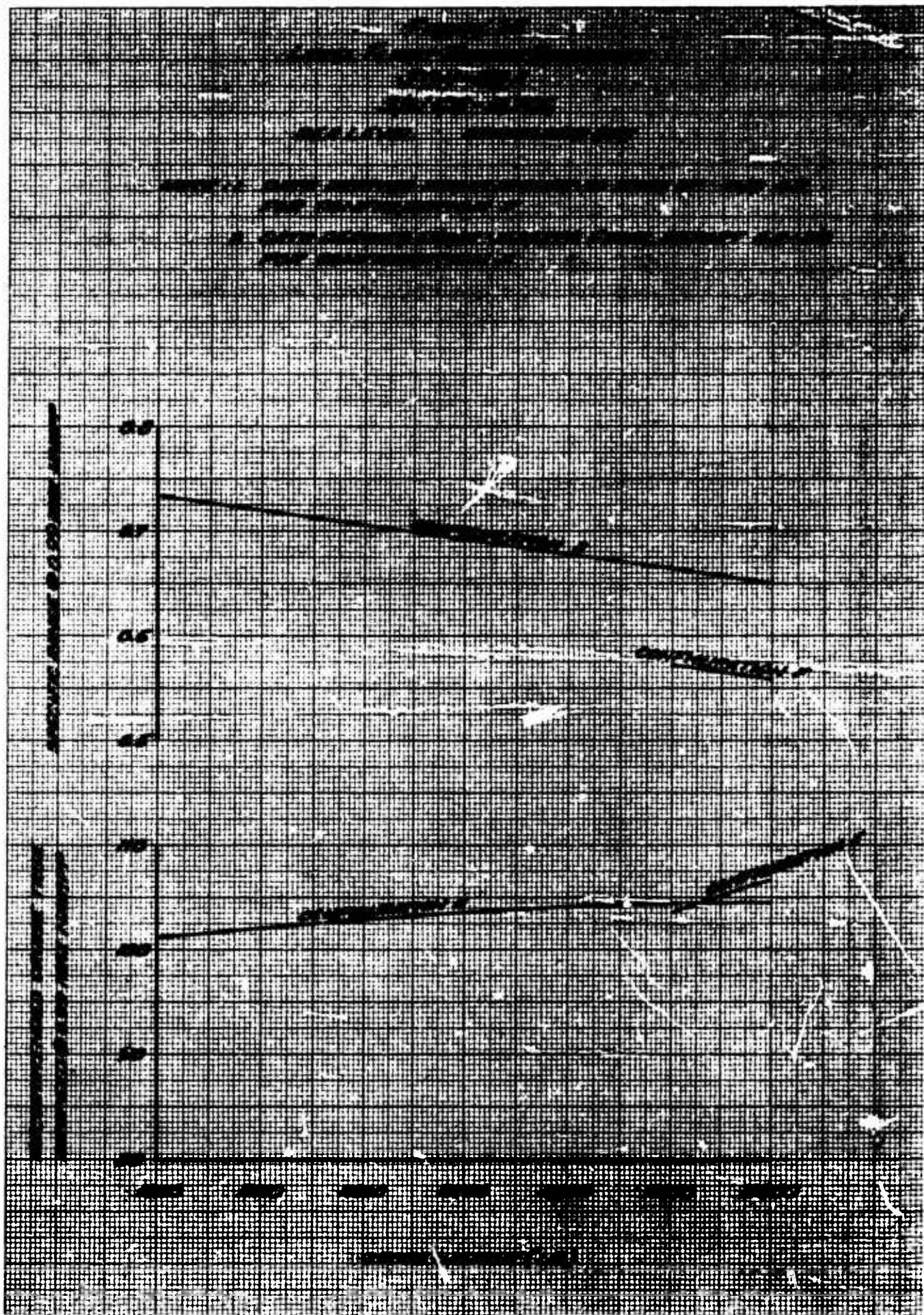
NOTE: 1. CONFIGURATION F DENOTES 12.5 IN. WING, 12.5 IN. STRIKE, STD. BODY  
 2. CURVES DERIVED FROM FIGS. 12 THRU 13





NOTE: 1. CALCULATIONS FOR  $\mu = 0.25, 0.30, 0.35, 0.40$   
 2. VALUES OF  $\mu$  ARE GIVEN IN THE TABLE

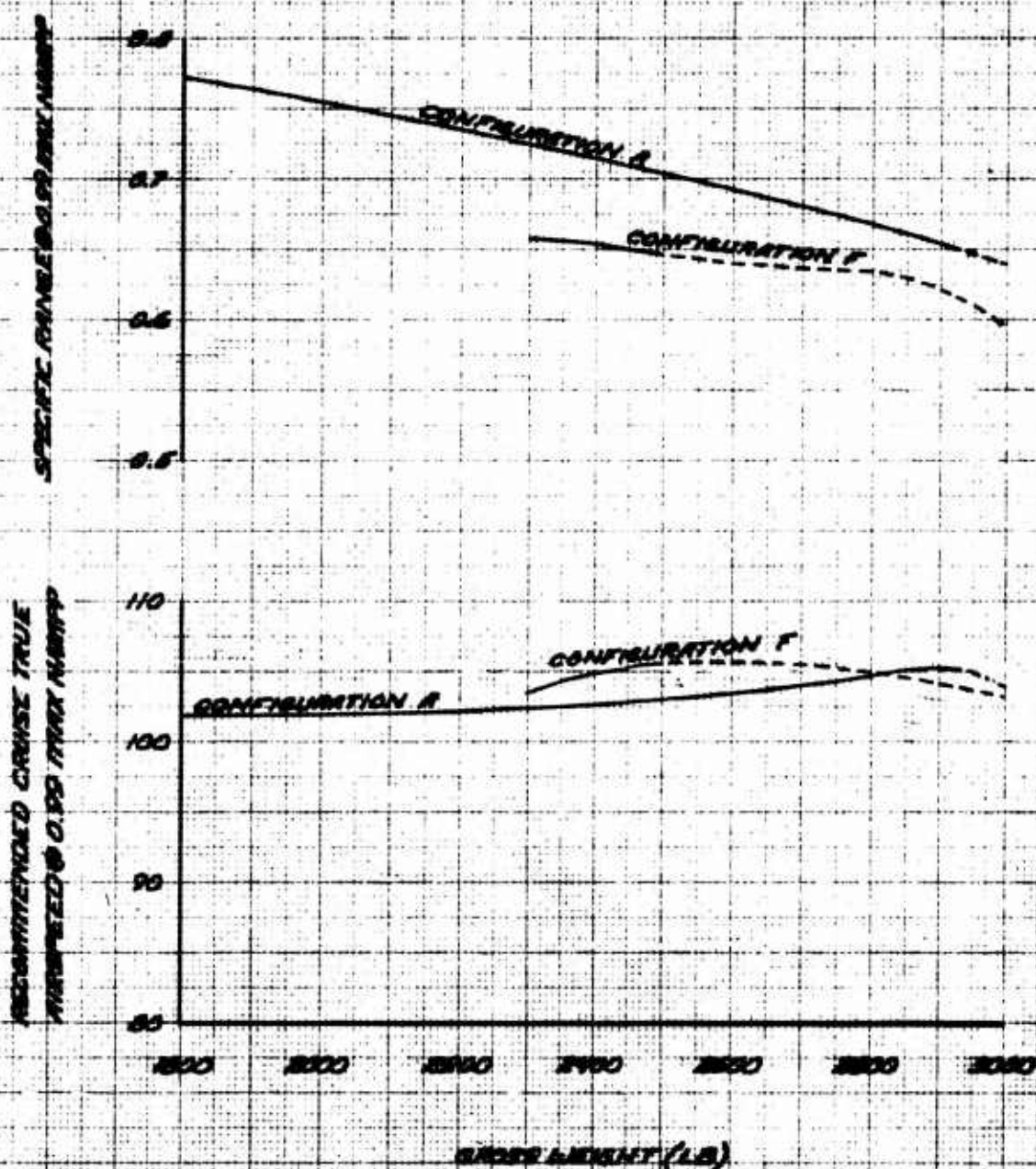


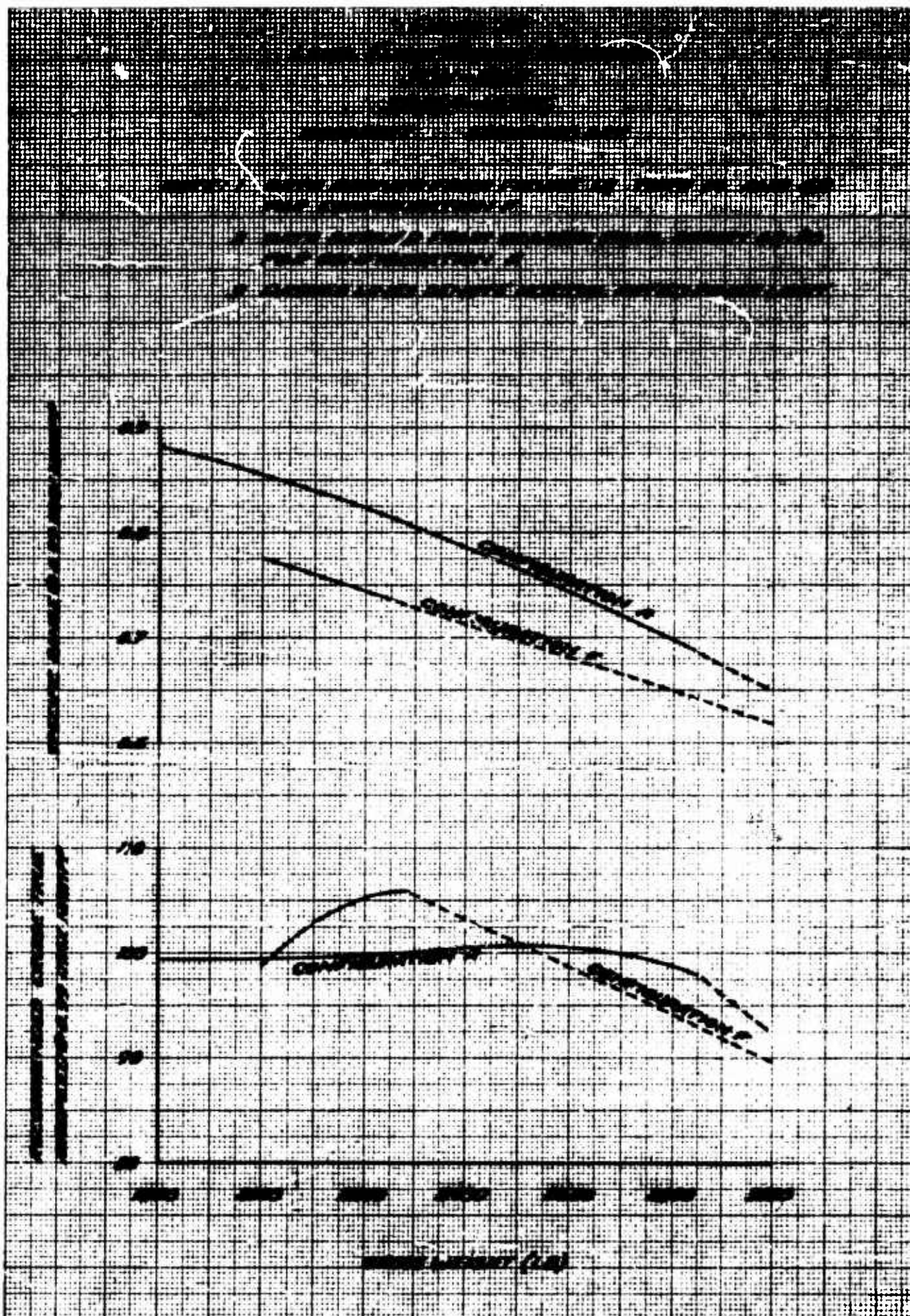




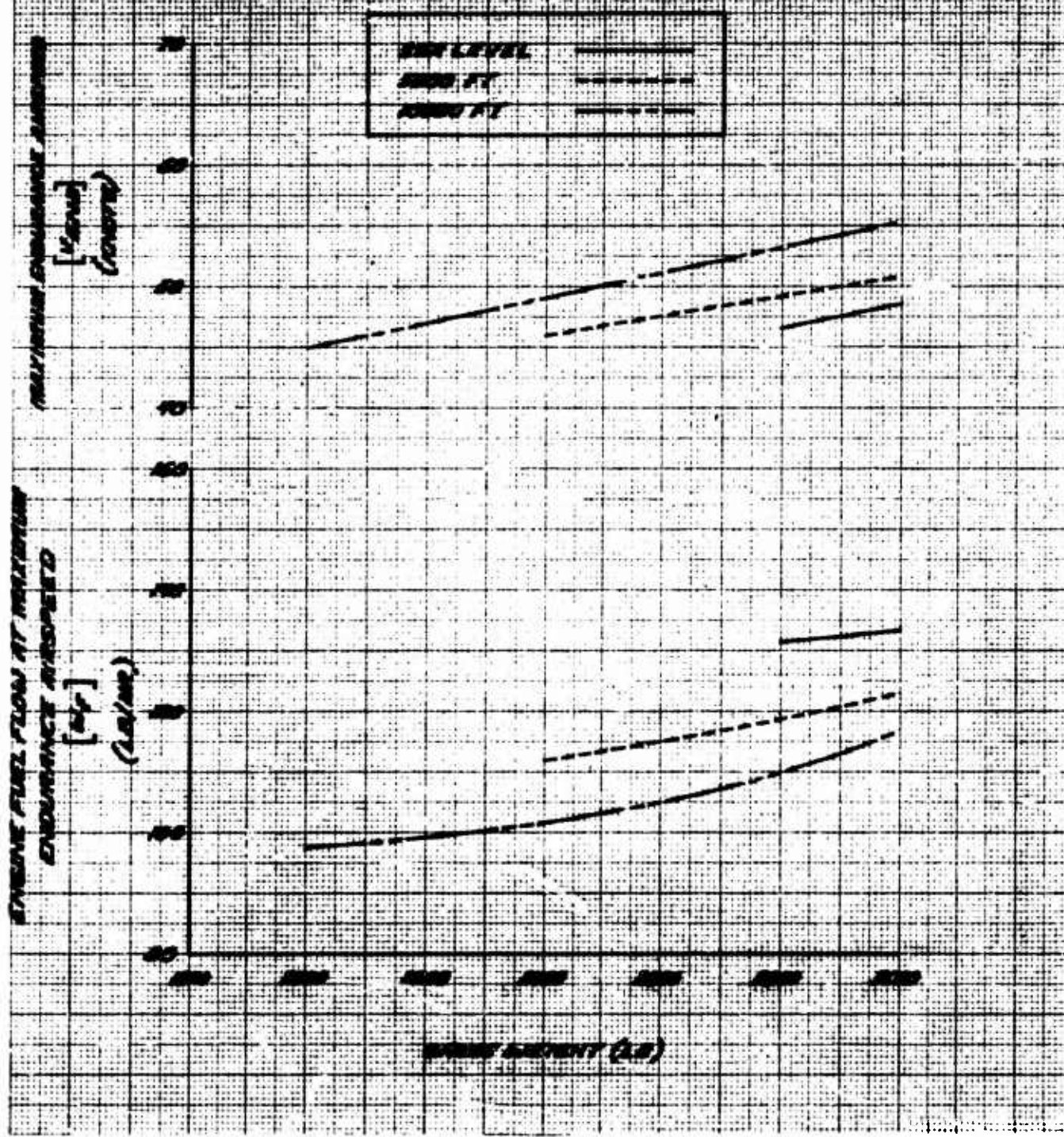
**Figure 12**  
**LEVEL FLIGHT ENGINE PERFORMANCE**  
**1000-2000**  
**5000-5500**  
**ENGINE: STANDARD 300**

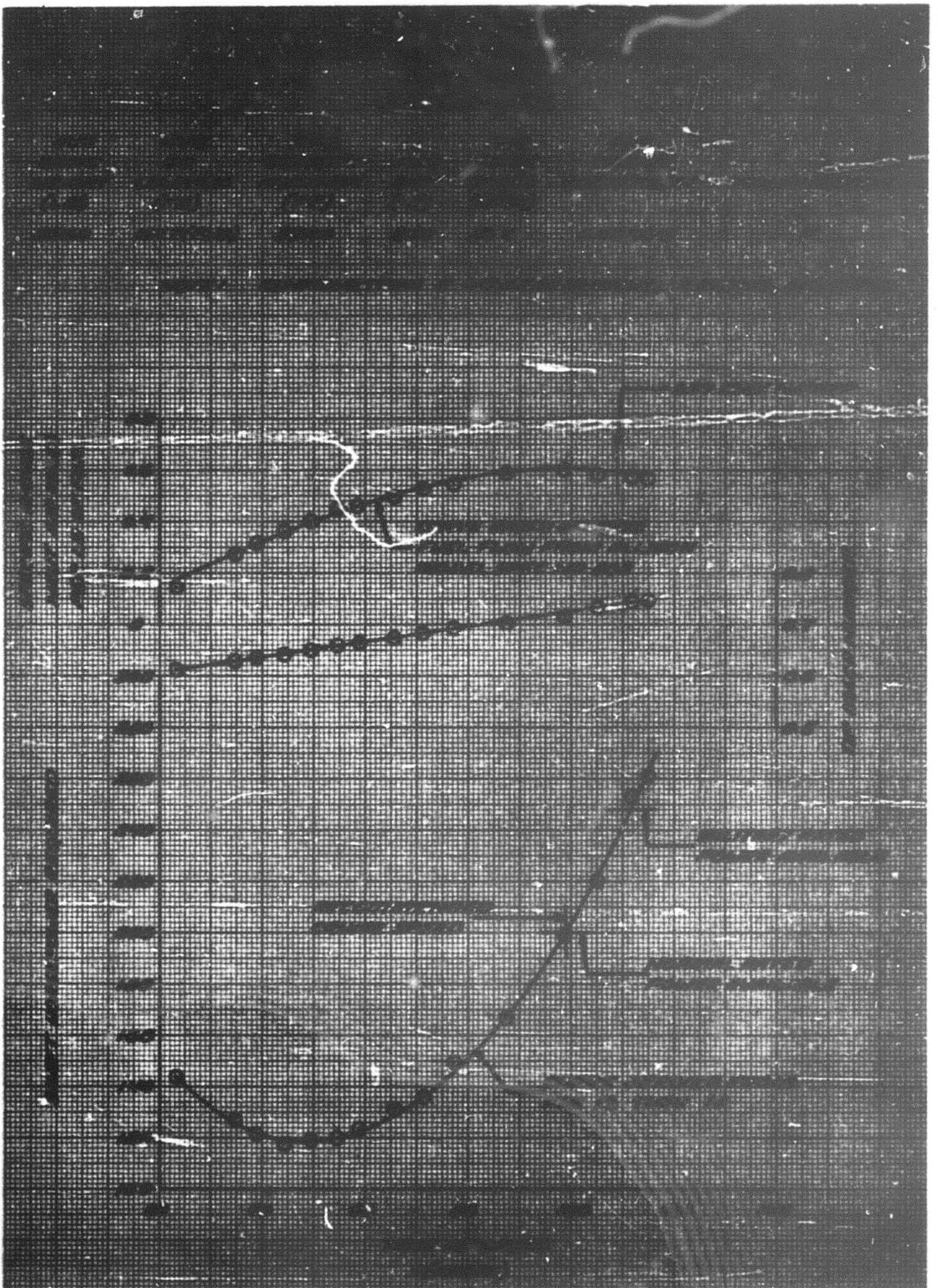
- NOTE 1:** DATA DERIVED FROM FIGURE 12 FROM 19 AND 20  
**FOR CONFIGURATION F**
- 2:** DATA DERIVED FROM LAMBERT PAUL REPORT 65-20  
**FOR CONFIGURATION A**
- 3:** DASHED LINES DENOTE MAXIMUM PERMITTED POWER LIMIT



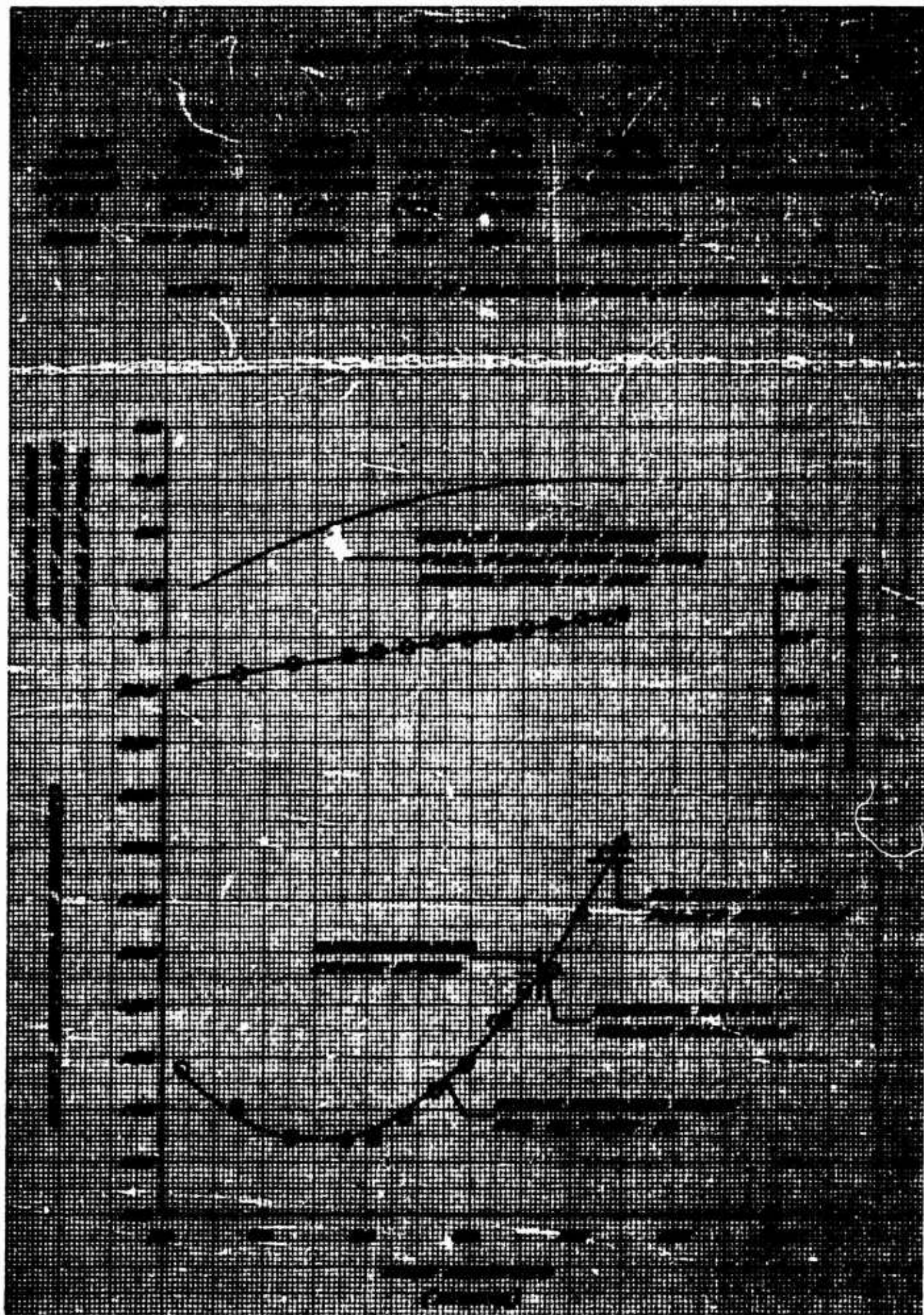


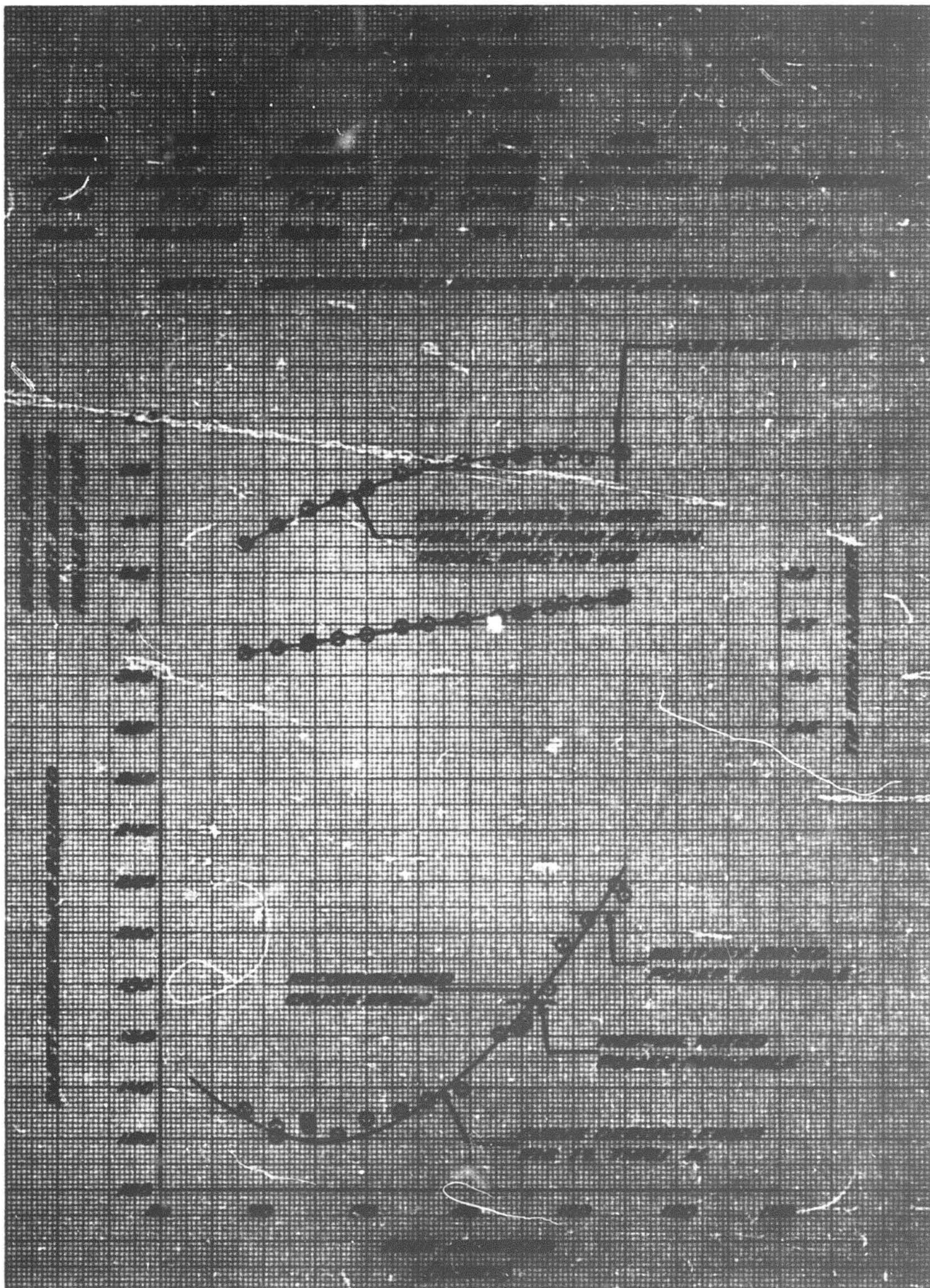




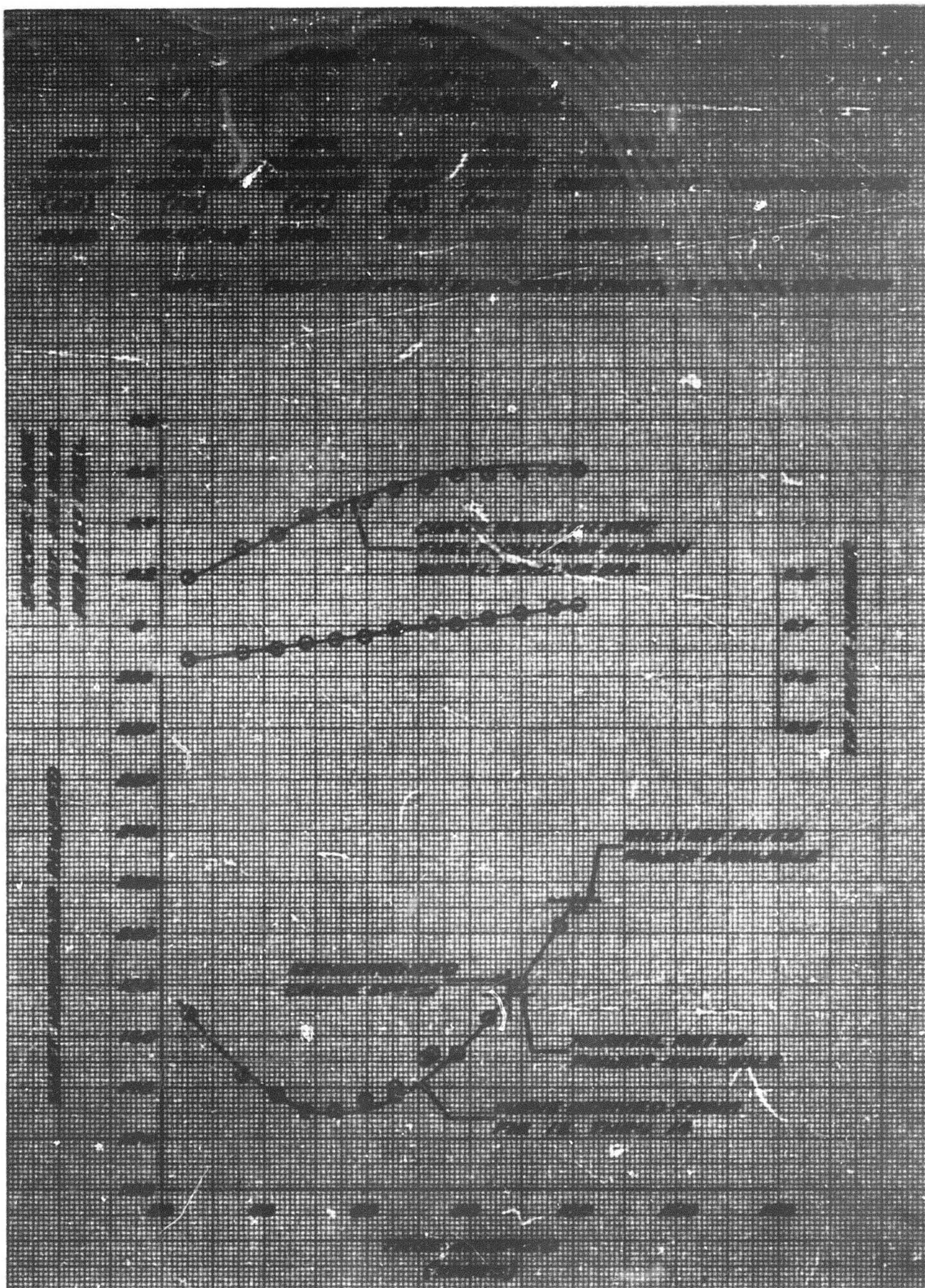




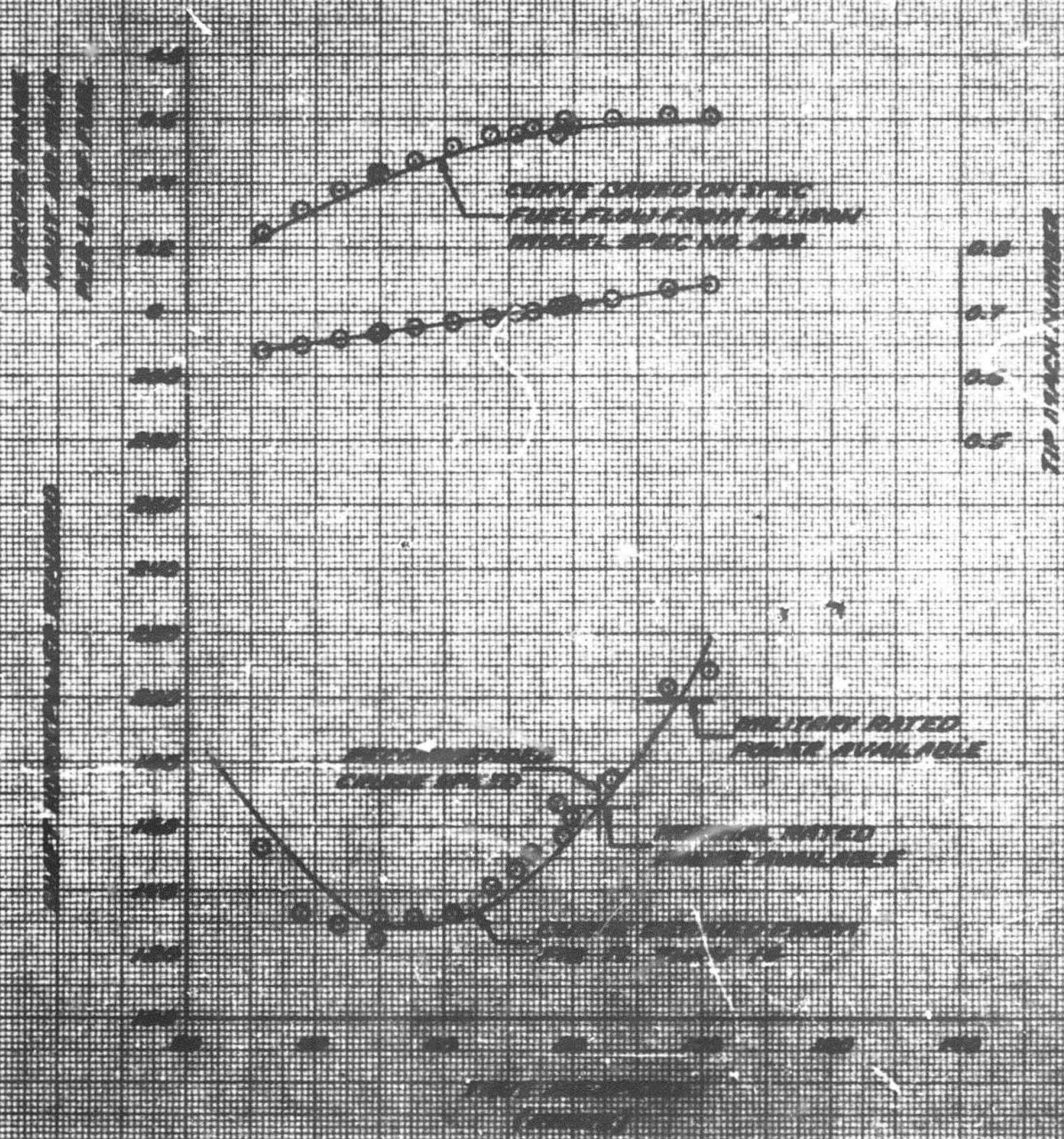








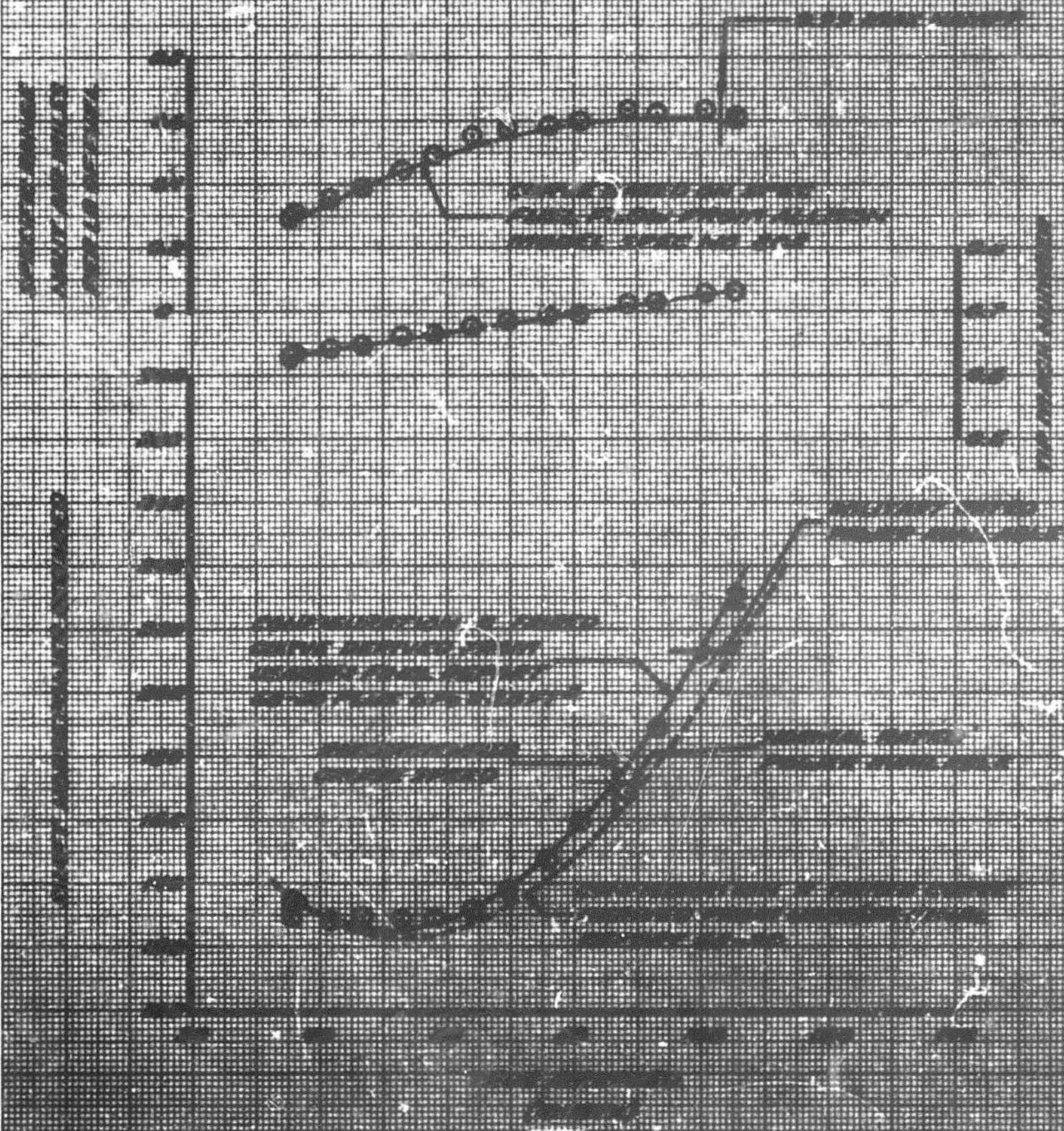
THE UNIVERSITY OF CHICAGO PRESS





TIME	TEMP	PH	DO	CO <sub>2</sub>	CHLOROPHYLL
0800	20.0	7.5	1.0	0.5	0.1
0900	20.5	7.5	1.0	0.5	0.1
1000	21.0	7.5	1.0	0.5	0.1
1100	21.5	7.5	1.0	0.5	0.1
1200	22.0	7.5	1.0	0.5	0.1
1300	22.5	7.5	1.0	0.5	0.1
1400	23.0	7.5	1.0	0.5	0.1
1500	23.5	7.5	1.0	0.5	0.1
1600	24.0	7.5	1.0	0.5	0.1
1700	24.5	7.5	1.0	0.5	0.1
1800	25.0	7.5	1.0	0.5	0.1
1900	25.5	7.5	1.0	0.5	0.1
2000	26.0	7.5	1.0	0.5	0.1
2100	26.5	7.5	1.0	0.5	0.1
2200	27.0	7.5	1.0	0.5	0.1
2300	27.5	7.5	1.0	0.5	0.1
0000	28.0	7.5	1.0	0.5	0.1
0100	28.5	7.5	1.0	0.5	0.1
0200	29.0	7.5	1.0	0.5	0.1
0300	29.5	7.5	1.0	0.5	0.1
0400	30.0	7.5	1.0	0.5	0.1
0500	30.5	7.5	1.0	0.5	0.1
0600	31.0	7.5	1.0	0.5	0.1
0700	31.5	7.5	1.0	0.5	0.1
0800	32.0	7.5	1.0	0.5	0.1
0900	32.5	7.5	1.0	0.5	0.1
1000	33.0	7.5	1.0	0.5	0.1
1100	33.5	7.5	1.0	0.5	0.1
1200	34.0	7.5	1.0	0.5	0.1
1300	34.5	7.5	1.0	0.5	0.1
1400	35.0	7.5	1.0	0.5	0.1
1500	35.5	7.5	1.0	0.5	0.1
1600	36.0	7.5	1.0	0.5	0.1
1700	36.5	7.5	1.0	0.5	0.1
1800	37.0	7.5	1.0	0.5	0.1
1900	37.5	7.5	1.0	0.5	0.1
2000	38.0	7.5	1.0	0.5	0.1
2100	38.5	7.5	1.0	0.5	0.1
2200	39.0	7.5	1.0	0.5	0.1
2300	39.5	7.5	1.0	0.5	0.1
0000	40.0	7.5	1.0	0.5	0.1
0100	40.5	7.5	1.0	0.5	0.1
0200	41.0	7.5	1.0	0.5	0.1
0300	41.5	7.5	1.0	0.5	0.1
0400	42.0	7.5	1.0	0.5	0.1
0500	42.5	7.5	1.0	0.5	0.1
0600	43.0	7.5	1.0	0.5	0.1
0700	43.5	7.5	1.0	0.5	0.1
0800	44.0	7.5	1.0	0.5	0.1
0900	44.5	7.5	1.0	0.5	0.1
1000	45.0	7.5	1.0	0.5	0.1
1100	45.5	7.5	1.0	0.5	0.1
1200	46.0	7.5	1.0	0.5	0.1
1300	46.5	7.5	1.0	0.5	0.1
1400	47.0	7.5	1.0	0.5	0.1
1500	47.5	7.5	1.0	0.5	0.1
1600	48.0	7.5	1.0	0.5	0.1
1700	48.5	7.5	1.0	0.5	0.1
1800	49.0	7.5	1.0	0.5	0.1
1900	49.5	7.5	1.0	0.5	0.1
2000	50.0	7.5	1.0	0.5	0.1
2100	50.5	7.5	1.0	0.5	0.1
2200	51.0	7.5	1.0	0.5	0.1
2300	51.5	7.5	1.0	0.5	0.1
0000	52.0	7.5	1.0	0.5	0.1
0100	52.5	7.5	1.0	0.5	0.1
0200	53.0	7.5	1.0	0.5	0.1
0300	53.5	7.5	1.0	0.5	0.1
0400	54.0	7.5	1.0	0.5	0.1
0500	54.5	7.5	1.0	0.5	0.1
0600	55.0	7.5	1.0	0.5	0.1
0700	55.5	7.5	1.0	0.5	0.1
0800	56.0	7.5	1.0	0.5	0.1
0900	56.5	7.5	1.0	0.5	0.1
1000	57.0	7.5	1.0	0.5	0.1
1100	57.5	7.5	1.0	0.5	0.1
1200	58.0	7.5	1.0	0.5	0.1
1300	58.5	7.5	1.0	0.5	0.1
1400	59.0	7.5	1.0	0.5	0.1
1500	59.5	7.5	1.0	0.5	0.1
1600	60.0	7.5	1.0	0.5	0.1
1700	60.5	7.5	1.0	0.5	0.1
1800	61.0	7.5	1.0	0.5	0.1
1900	61.5	7.5	1.0	0.5	0.1
2000	62.0	7.5	1.0	0.5	0.1
2100	62.5	7.5	1.0	0.5	0.1
2200	63.0	7.5	1.0	0.5	0.1
2300	63.5	7.5	1.0	0.5	0.1
0000	64.0	7.5	1.0	0.5	0.1
0100	64.5	7.5	1.0	0.5	0.1
0200	65.0	7.5	1.0	0.5	0.1
0300	65.5	7.5	1.0	0.5	0.1
0400	66.0	7.5	1.0	0.5	0.1
0500	66.5	7.5	1.0	0.5	0.1
0600	67.0	7.5	1.0	0.5	0.1
0700	67.5	7.5	1.0	0.5	0.1
0800	68.0	7.5	1.0	0.5	0.1
0900	68.5	7.5	1.0	0.5	0.1
1000	69.0	7.5	1.0	0.5	0.1
1100	69.5	7.5	1.0	0.5	0.1
1200	70.0	7.5	1.0	0.5	0.1
1300	70.5	7.5	1.0	0.5	0.1
1400	71.0	7.5	1.0	0.5	0.1
1500	71.5	7.5	1.0	0.5	0.1
1600	72.0	7.5	1.0	0.5	0.1
1700	72.5	7.5	1.0	0.5	0.1
1800	73.0	7.5	1.0	0.5	0.1
1900	73.5	7.5	1.0	0.5	0.1
2000	74.0	7.5	1.0	0.5	0.1
2100	74.5	7.5	1.0	0.5	0.1
2200	75.0	7.5	1.0	0.5	0.1
2300	75.5	7.5	1.0	0.5	0.1
0000	76.0	7.5	1.0	0.5	0.1
0100	76.5	7.5	1.0	0.5	0.1
0200	77.0	7.5	1.0	0.5	0.1
0300	77.5	7.5	1.0	0.5	0.1
0400	78.0	7.5	1.0	0.5	0.1
0500	78.5	7.5	1.0	0.5	0.1
0600	79.0	7.5	1.0	0.5	0.1
0700	79.5	7.5	1.0	0.5	0.1
0800	80.0	7.5	1.0	0.5	0.1
0900	80.5	7.5	1.0	0.5	0.1
1000	81.0	7.5	1.0	0.5	0.1
1100	81.5	7.5	1.0	0.5	0.1
1200	82.0	7.5	1.0	0.5	0.1
1300	82.5	7.5	1.0	0.5	0.1
1400	83.0	7.5	1.0	0.5	0.1
1500	83.5	7.5	1.0	0.5	0.1
1600	84.0	7.5	1.0	0.5	0.1
1700	84.5	7.5	1.0	0.5	0.1
1800	85.0	7.5	1.0	0.5	0.1
1900	85.5	7.5	1.0	0.5	0.1
2000	86.0	7.5	1.0	0.5	0.1
2100	86.5	7.5	1.0	0.5	0.1
2200	87.0	7.5	1.0	0.5	0.1
2300	87.5	7.5	1.0	0.5	0.1
0000	88.0	7.5	1.0	0.5	0.1
0100	88.5	7.5	1.0	0.5	0.1
0200	89.0	7.5	1.0	0.5	0.1
0300	89.5	7.5	1.0	0.5	0.1
0400	90.0	7.5	1.0	0.5	0.1
0500	90.5	7.5	1.0	0.5	0.1
0600	91.0	7.5	1.0	0.5	0.1
0700	91.5	7.5	1.0	0.5	0.1
0800	92.0	7.5	1.0	0.5	0.1
0900	92.5	7.5	1.0	0.5	0.1
1000	93.0	7.5	1.0	0.5	0.1
1100	93.5	7.5	1.0	0.5	0.1
1200	94.0	7.5	1.0	0.5	0.1
1300	94.5	7.5	1.0	0.5	0.1
1400	95.0	7.5	1.0	0.5	0.1
1500	95.5	7.5	1.0	0.5	0.1
1600	96.0	7.5	1.0	0.5	0.1
1700	96.5	7.5	1.0	0.5	0.1
1800	97.0	7.5	1.0	0.5	0.1
1900	97.5	7.5	1.0	0.5	0.1
2000	98.0	7.5	1.0	0.5	0.1
2100	98.5	7.5	1.0	0.5	0.1
2200	99.0	7.5	1.0	0.5	0.1
2300	99.5	7.5	1.0	0.5	0.1
0000	100.0	7.5	1.0	0.5	0.1

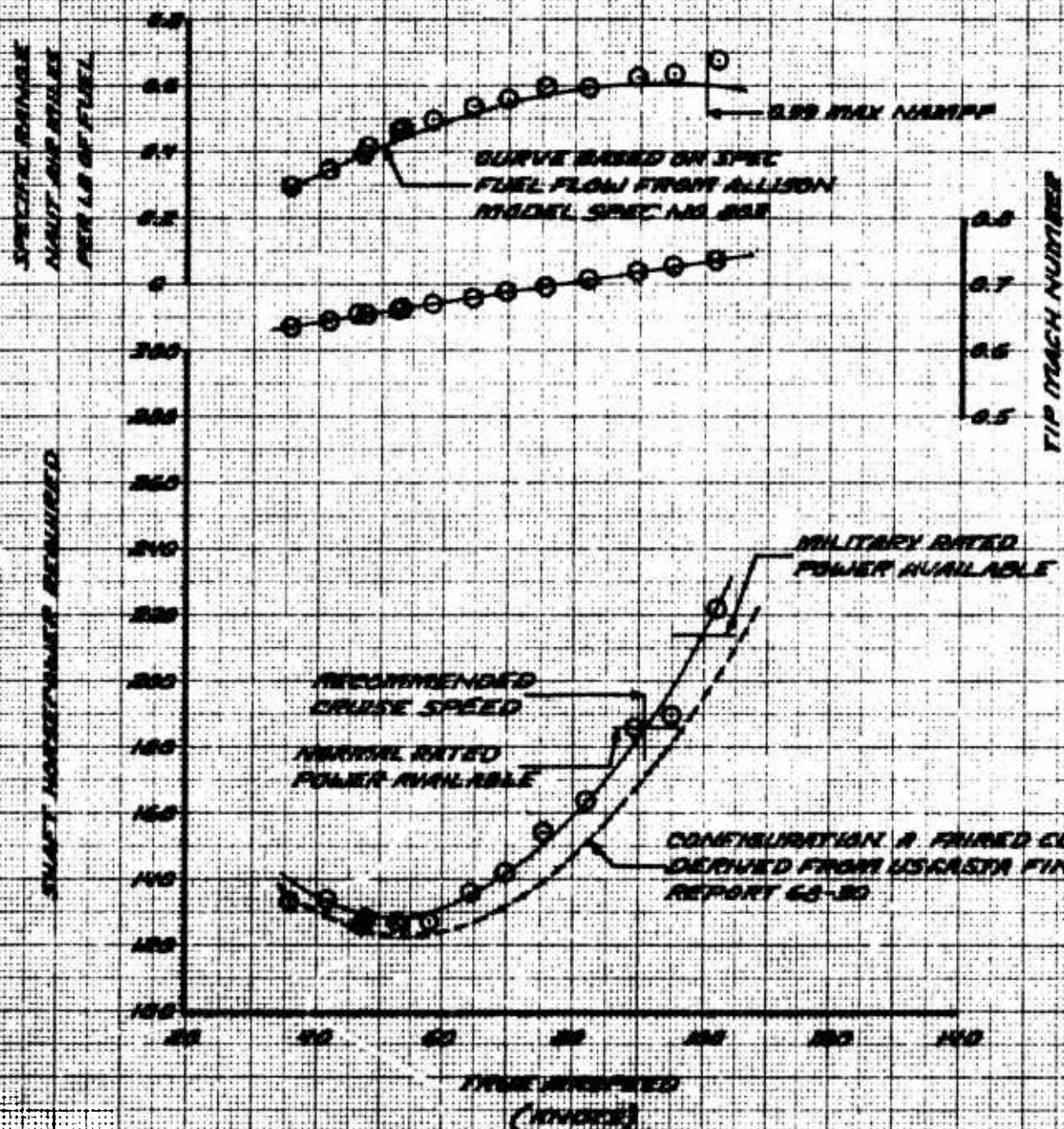
NOTE: TEMPERATURES & DO VALUES ARE FROM THE SURFACE OF THE WATER.



**Figure 25**  
**Engine Flight Performance**  
**YF-12**  
**YF-12A**

YF-12	YF-12A	YF-12	YF-12A	YF-12	YF-12A	
WEIGHT	WEIGHT	WEIGHT	WEIGHT	WEIGHT	WEIGHT	
(LBS)	(LBS)	(LBS)	(LBS)	(LBS)	(LBS)	
27,000	27,000	27,000	27,000	27,000	27,000	

MODEL 1 (1) AND MODEL 2 (2) ENGINE 12 POINT, 12 STACKS, 12 BURNERS

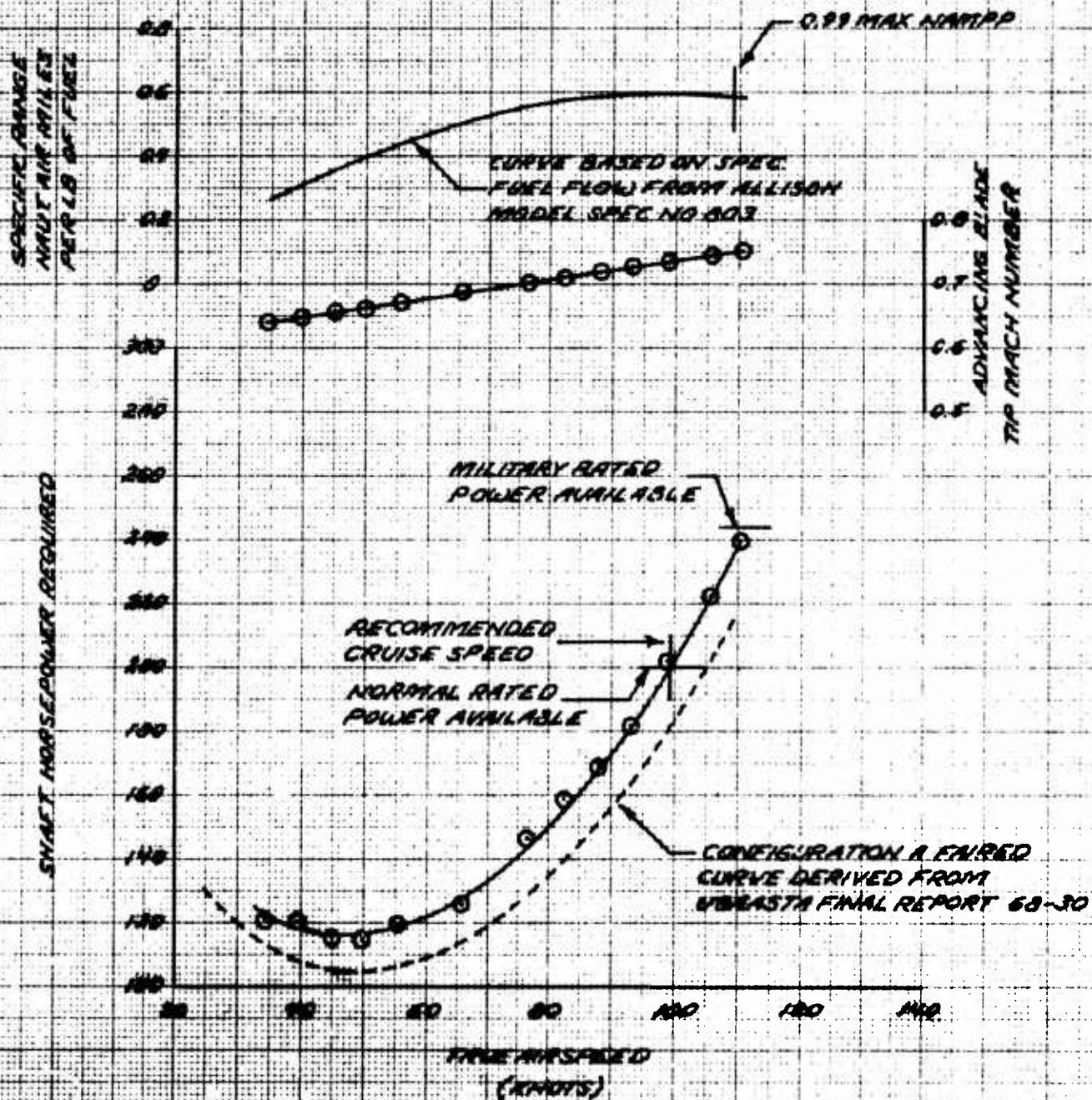




**Figure 106  
LAHEL Flight Performance  
JON-50A  
51450-16706**

AVG WEIGHT (LB)	AVG CG LOCATION (IN)	AVG DENSITY ALTITUDE (FT)	AVG ROTOR DRIFT SPEED (MPH)	AVG THRUST EFFICIENCY	CONFIGURATION
2950	102.5000	4420	28.8	854	0.068004
					E

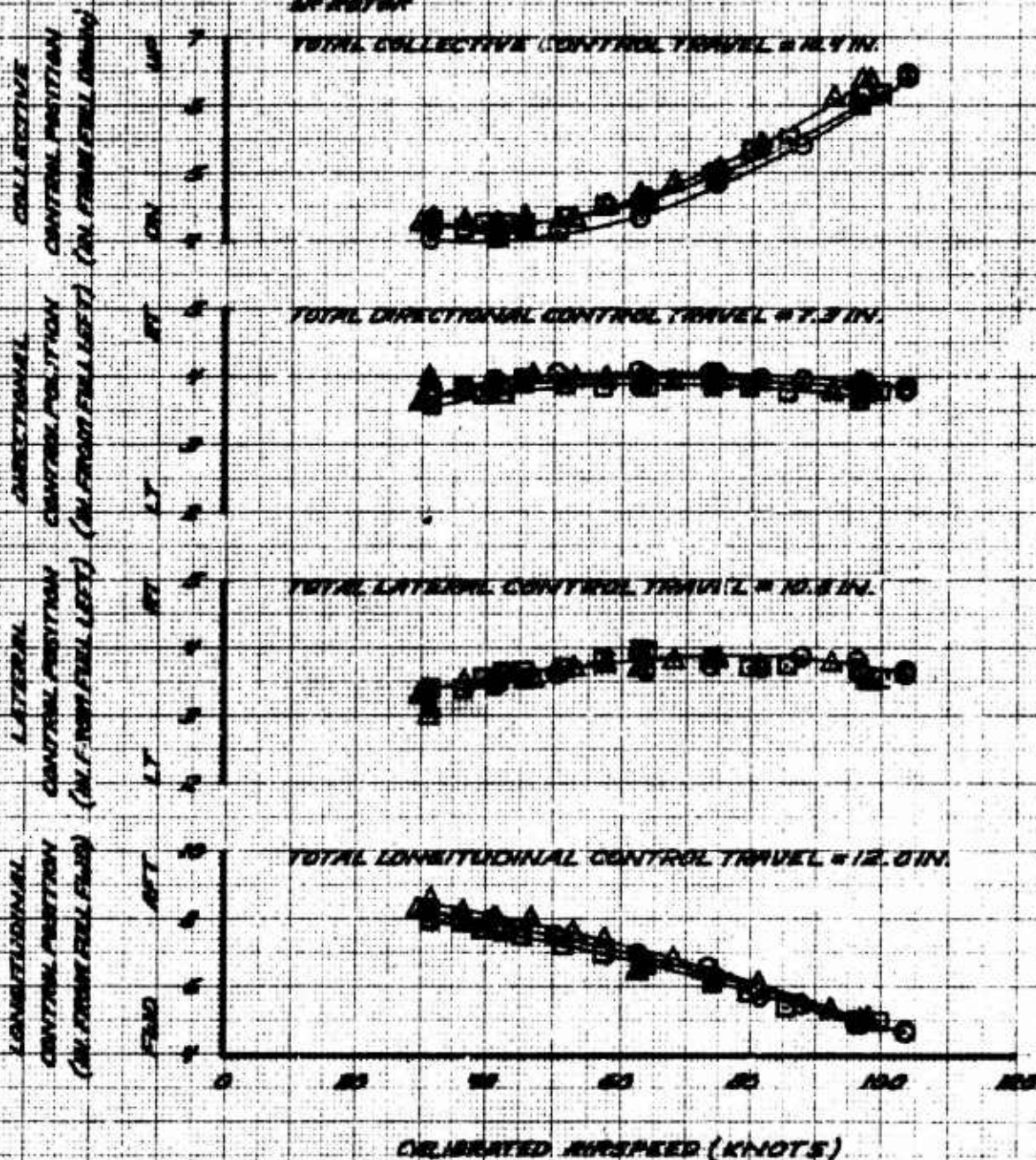
NOTE: CONFIGURATION E DENOTES LR PAINT, LR STRIPS, MOUNTED  
LR ROTOR



**FIGURE 27**  
**Control Parameters Summary**  
**JCM-522**  
**S/N 65-16706**

TYPE	AVG WEIGHT (LB)	AVG OR LENGTH (IN)	AVG DENSITY ALTITUDE (FT)	AVG OUT (IN)	AVG SPEED (RPM)	AVG CY	CONFIGURATION
○	2500	107.5 (240)	7200	170	354	0.00000	A
□	2500	107.5 (240)	7200	20.2	354	0.00000	B
△	2500	107.5 (240)	7200	20.5	354	0.00002	C

NOTE: 1. CONFIGURATION A DENOTES STD PRINT, STD STACKS, STD ROTOR  
 2. CONFIGURATION B DENOTES STD PRINT, 1R STACKS, STD ROTOR  
 3. CONFIGURATION C DENOTES DR PRINT, 1R STACKS, MODIFIED  
 4R ROTOR

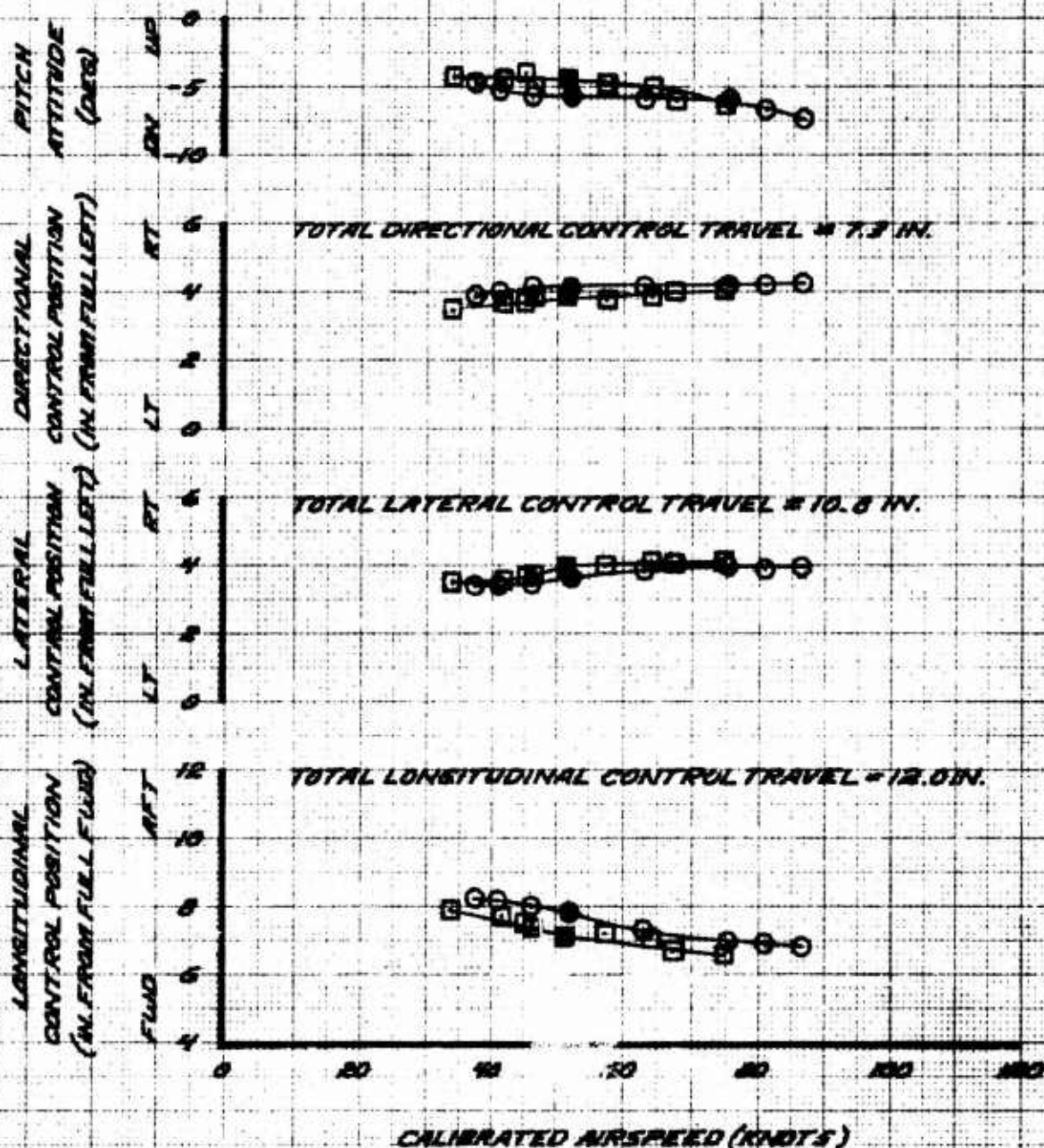




**Figure 25**  
**Collective - Fixed Static Longitudinal Stability**  
**JON-58A**  
**SINGA-K706**

SYM	Avg Gross Weight (LB)	Avg CG Location (IN)	Avg Density Altitude (FT)	Avg GAT (%)	Avg RPM	Configuration
○	2850	MT.8(FWD)	6500	84.5	254	A
□	2900	MT.9(FWD)	6500	84.5	254	B

NOTE: 1 SHADED SYMBOL DENOTES TRIM  
 2 CONFIGURATION A DENOTES STD PRNT, STD STKS, STD RSTR  
 3 CONFIGURATION B DENOTES STD PRNT, 12 STKS, STD RSTR





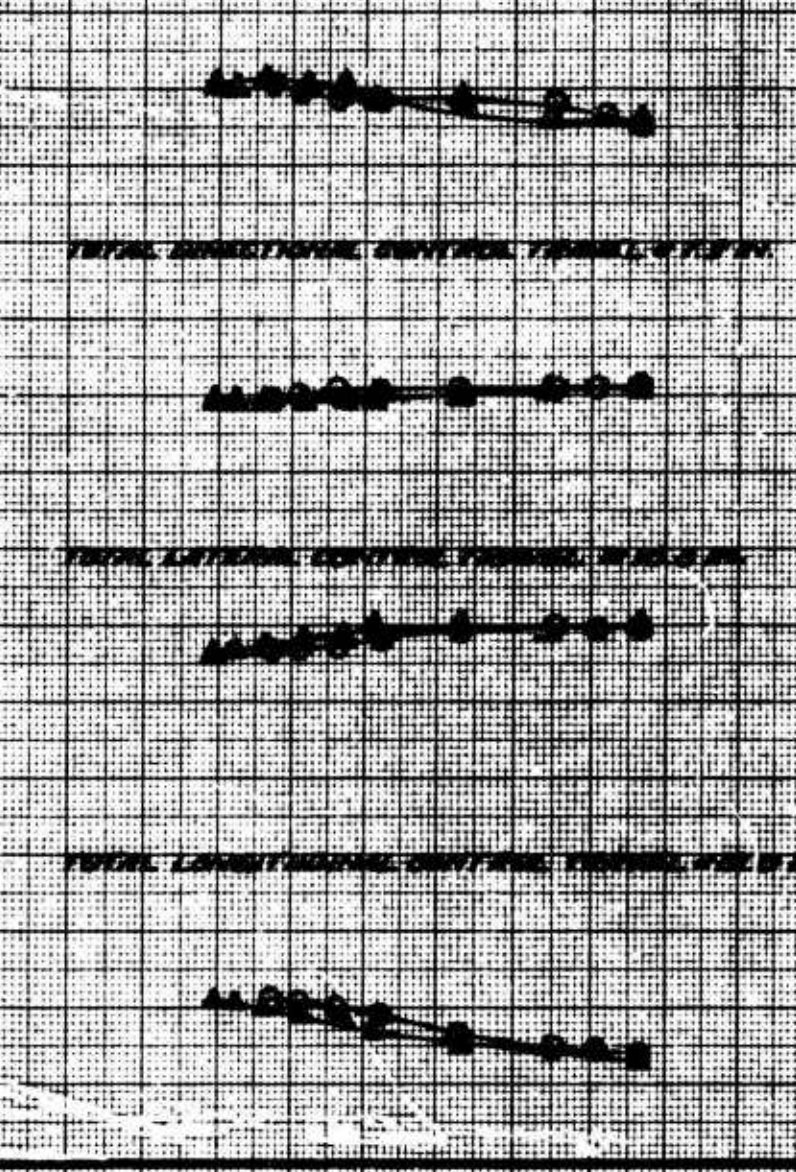


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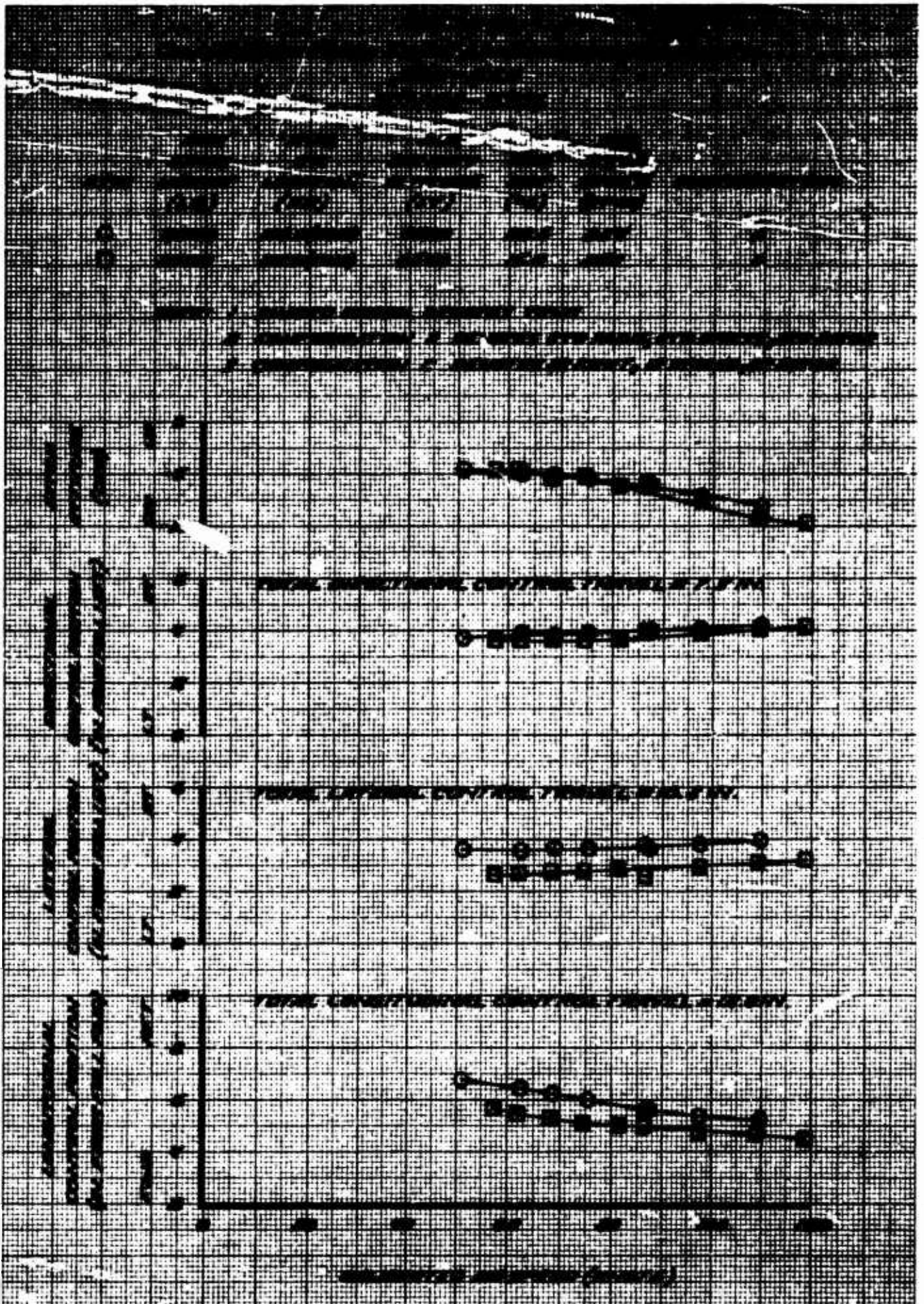
1. *...*  
 2. *...*  
 3. *...*  
 4. *...*  
 5. *...*  
 6. *...*  
 7. *...*  
 8. *...*  
 9. *...*  
 10. *...*

1. *...*  
 2. *...*  
 3. *...*  
 4. *...*  
 5. *...*  
 6. *...*  
 7. *...*  
 8. *...*  
 9. *...*  
 10. *...*

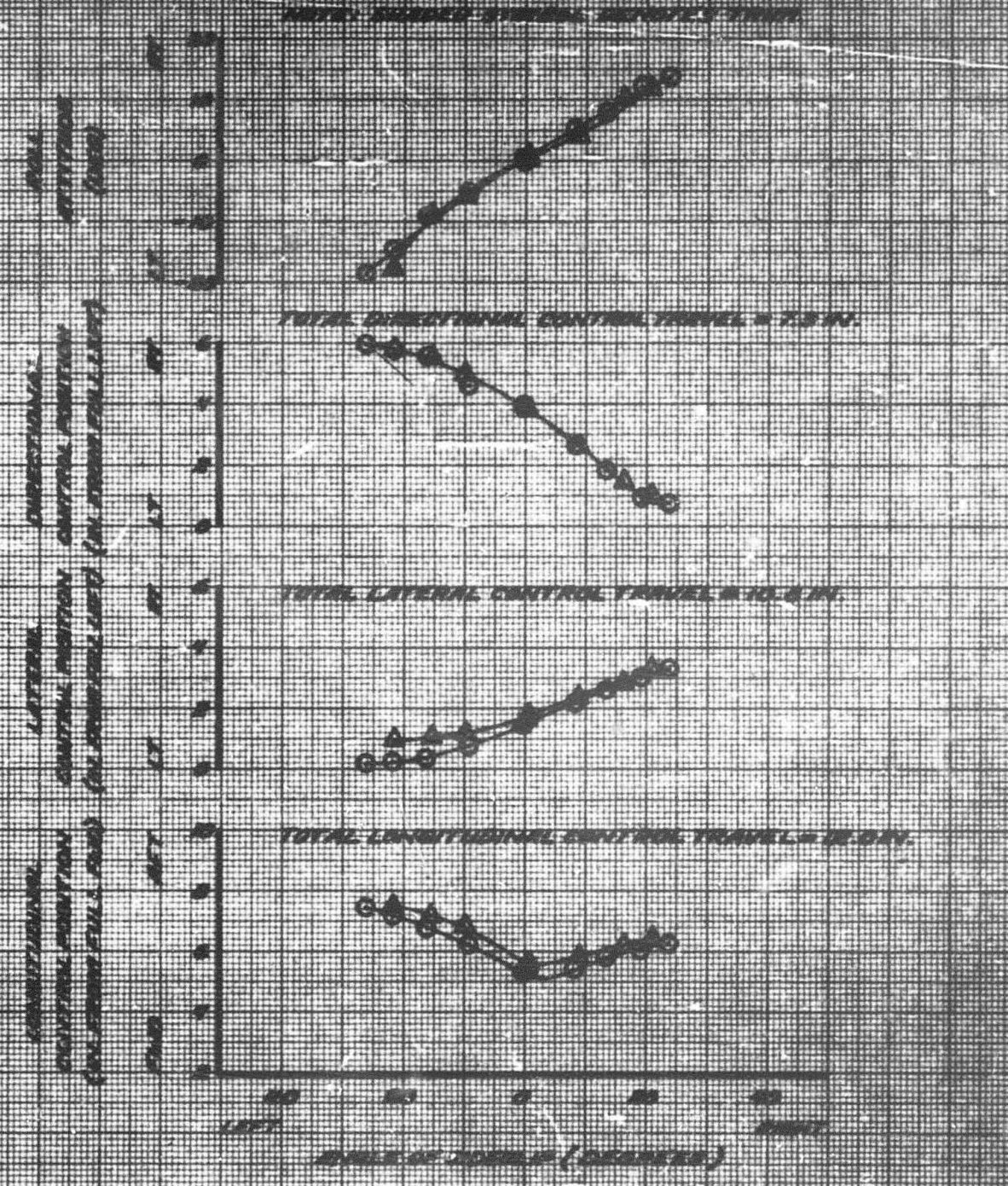
1. *...*  
 2. *...*  
 3. *...*  
 4. *...*  
 5. *...*  
 6. *...*  
 7. *...*  
 8. *...*  
 9. *...*  
 10. *...*



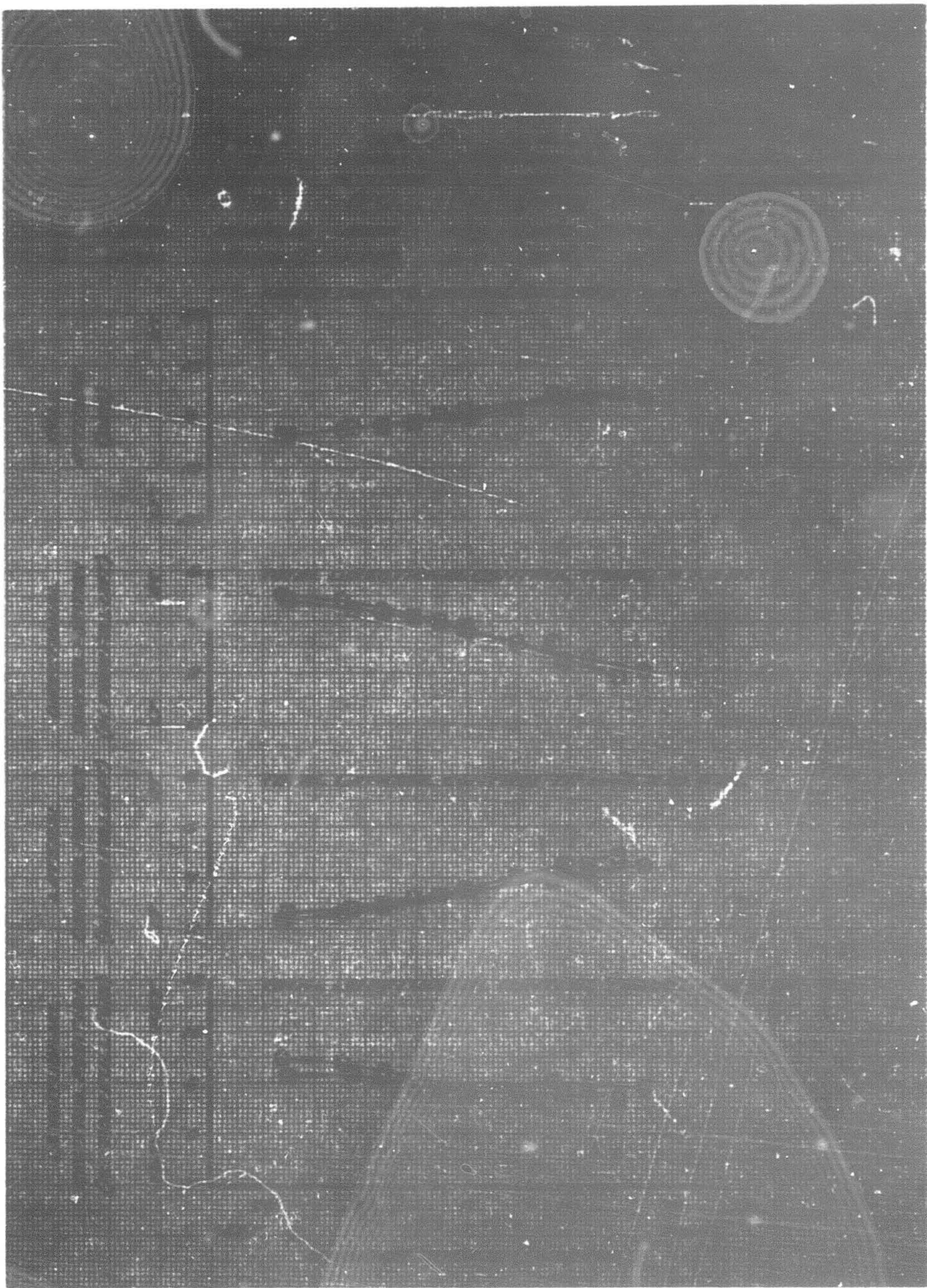




NO.	DATE	TIME	WIND	TEMP	REL. HUM.	SEA
1	10/10/54	10:00	10	60	70	1
2	10/10/54	11:00	10	60	70	1
3	10/10/54	12:00	10	60	70	1
4	10/10/54	13:00	10	60	70	1
5	10/10/54	14:00	10	60	70	1
6	10/10/54	15:00	10	60	70	1
7	10/10/54	16:00	10	60	70	1
8	10/10/54	17:00	10	60	70	1
9	10/10/54	18:00	10	60	70	1
10	10/10/54	19:00	10	60	70	1



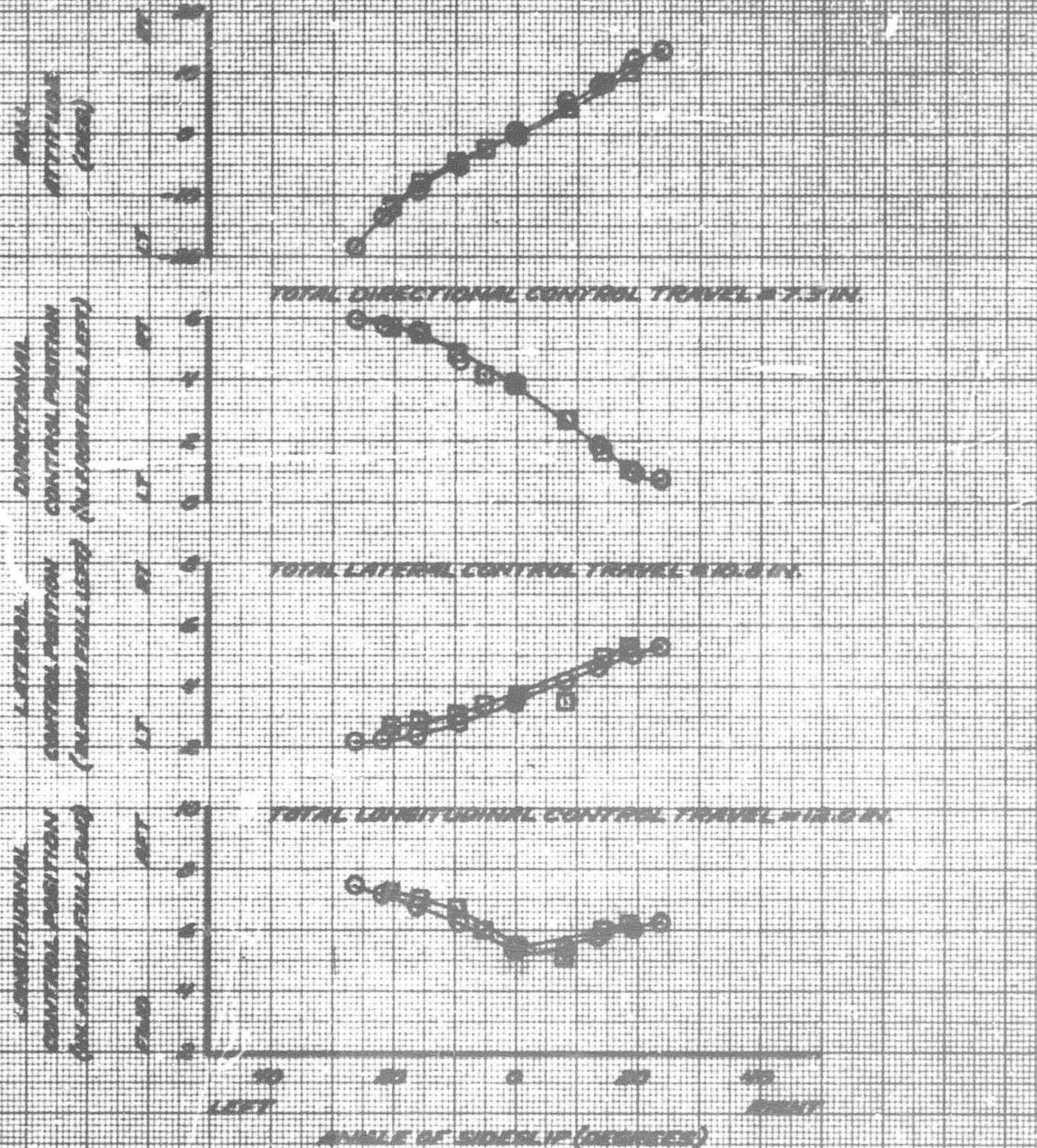




# Figure 30 Static Lateral-Directional Stability JON-SAS C-145-AC06

TYPE	TYPE	TYPE	TYPE	TYPE	TYPE	
WING	WING	WING	WING	WING	WING	
WING	WING	WING	WING	WING	WING	
(1.0)	(1.0)	(1.0)	(1.0)	(1.0)	(1.0)	
WING	WING	WING	WING	WING	WING	
WING	WING	WING	WING	WING	WING	

WING-ENHANCED STABILITY DERIVATIVES

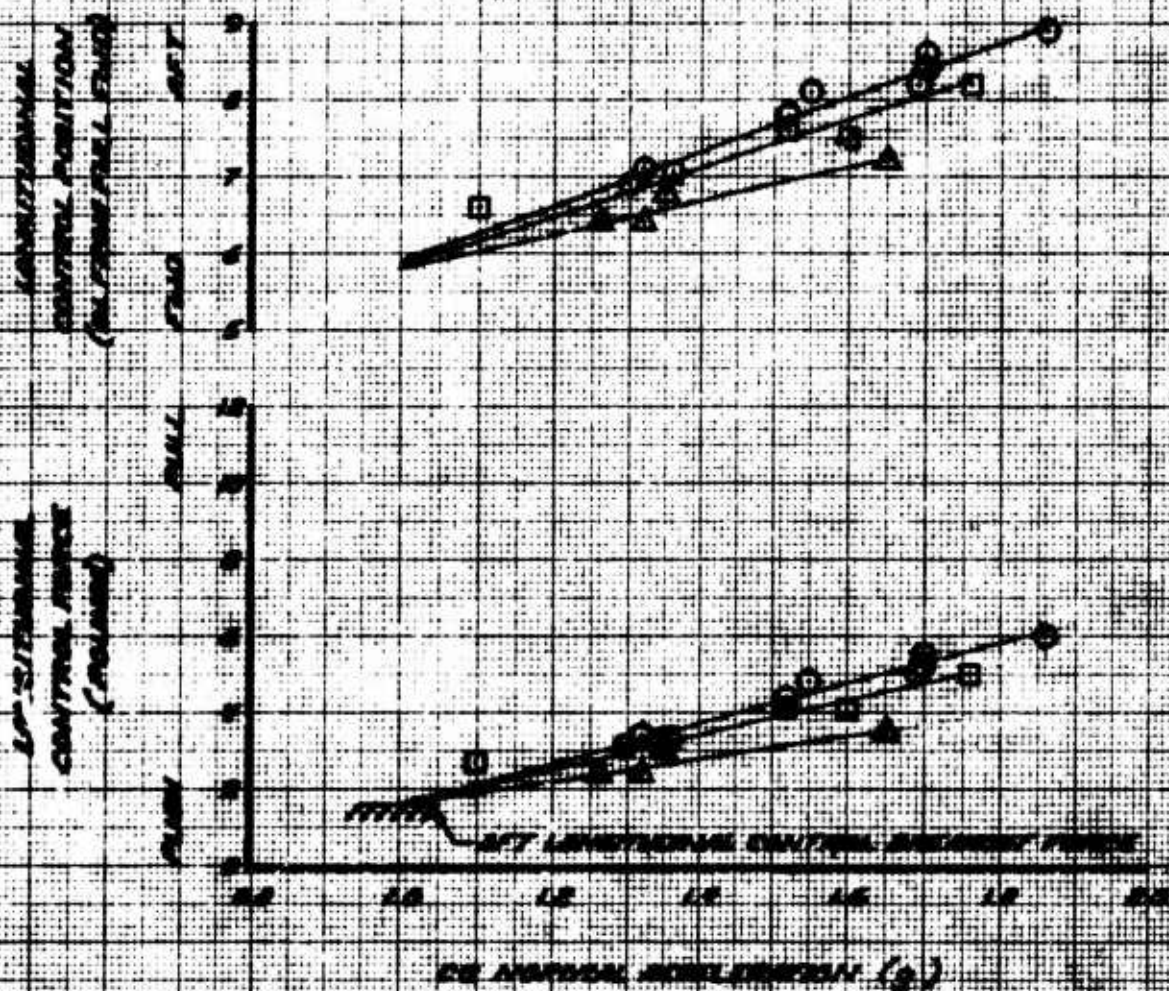




**Figure 10**  
**Relationships of Control**  
**Variables**  
**Signal-Noise**

Variable	Control Variable	Control Variable	Control Variable	Control Variable	Control Variable	Control Variable
Variable	Control Variable	Control Variable	Control Variable	Control Variable	Control Variable	Control Variable
C	Control	Control	Control	Control	Control	Control
D	Control	Control	Control	Control	Control	Control
A	Control	Control	Control	Control	Control	Control

NOTE: 1. CONTINUATION A. DETECTS ITS POINT, ITS POINT, ITS POINT  
2. CONTINUATION B. DETECTS ITS POINT, ITS POINT, ITS POINT  
3. CONTINUATION C. DETECTS ITS POINT, ITS POINT, ITS POINT





# FIGURE 1 LONGITUDINAL LONG TERM RESPONSE

MODEL 103-A-700 ENGINE

GROSS WEIGHT - LB 2000  
CS LOCATION - IN 107.8 (FWD)  
DENSITY ALTITUDE - FT 7660  
WAT - DEG C 24.0  
ROTOR SPEED - RPM 364  
CALIBRATED AIRSPEED - KTS 62  
THRUST COEFFICIENT 0.00582  
CONFIGURATION CONFIG. A

NOTE: SCNS OFF

SOLID LINE  
SHORT DASH  
LONG DASH

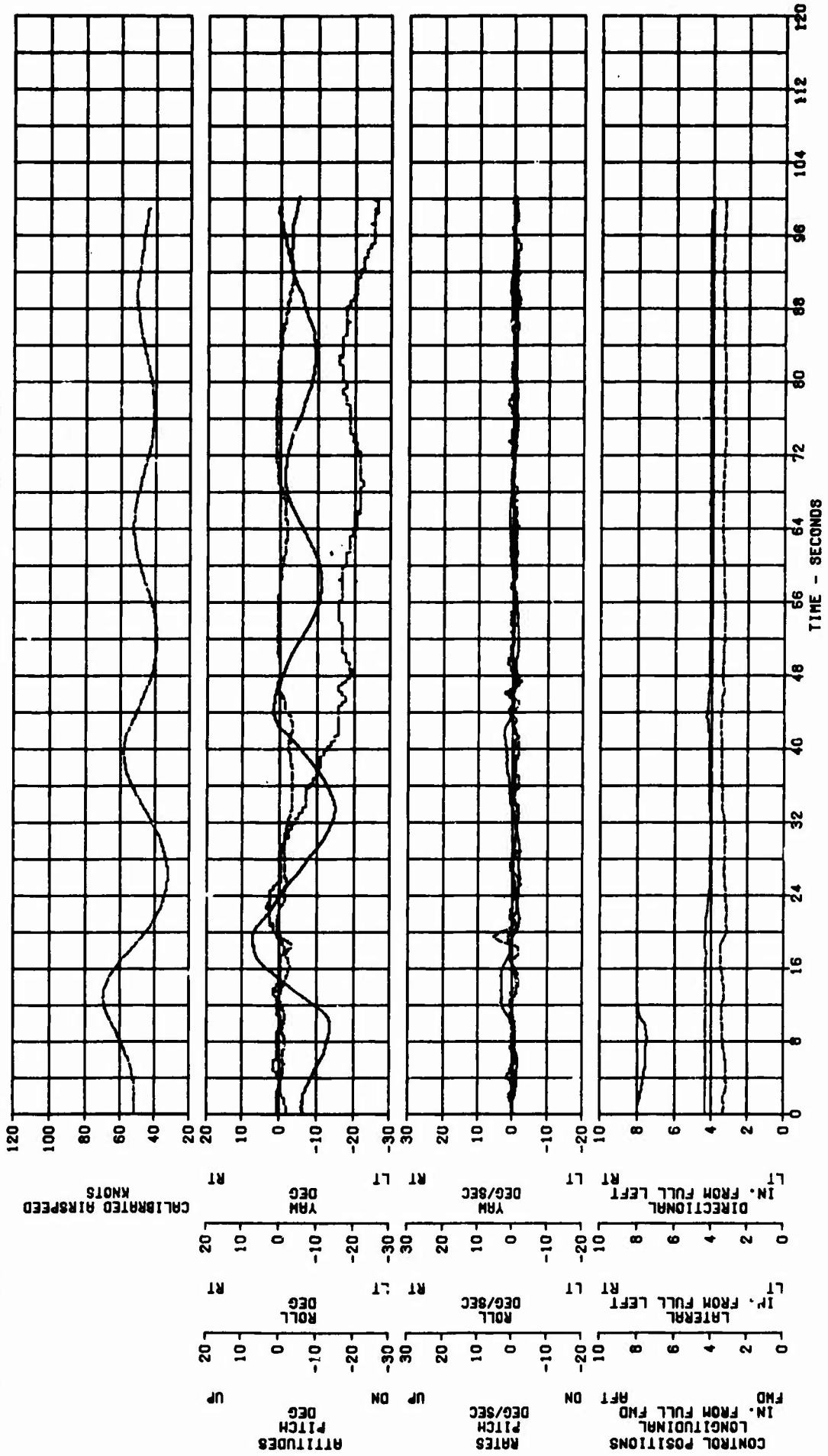
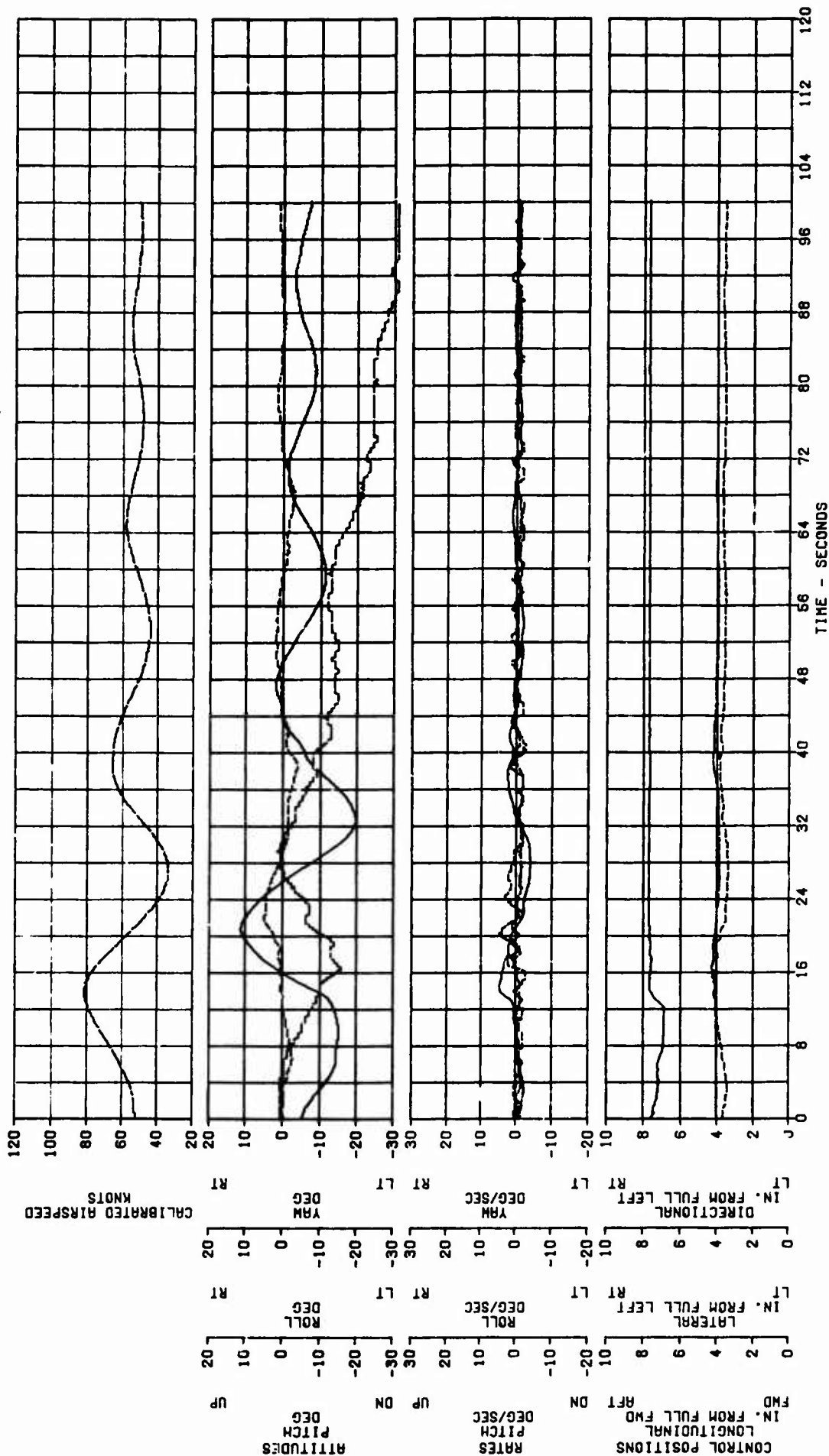


FIGURE 37  
LONGITUDINAL LONG TERM RESPONSE  
JAN-588 B/A 88-10508  
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB 2800  
CO LOCATION - IN 107.8 (FWD)  
DENSITY ALTITUDE - FT 7600  
OAT - DEG C 24.0  
ROTOR SPEED - RPM 354  
CALIBRATED AIRSPEED - KTS 62  
THRUST COEFFICIENT 0.00362  
CONFIGURATION CONFIG. A

NOTE: SCRS OFF

SOLID LINE SHORT DASH LONG DASH



# Figure 38 LONGITUDINAL LONG TERM RESPONSE

MODEL: F3-4-700-20012  
 GROSS WEIGHT - LB 2700  
 CG LOCATION - IN 107.6 (FM0)  
 DENSITY ALTITUDE - FT 7400  
 GWT - DEG L 23.5  
 ROTOR FREED - RPM 364  
 CALIBRATED AIRSPEED - KTS 0  
 THRUST COEFFICIENT 0.00348  
 CONFIG - A

NOTE: SCAS OFF

SOLID LINE SHORT DASH LONG DASH

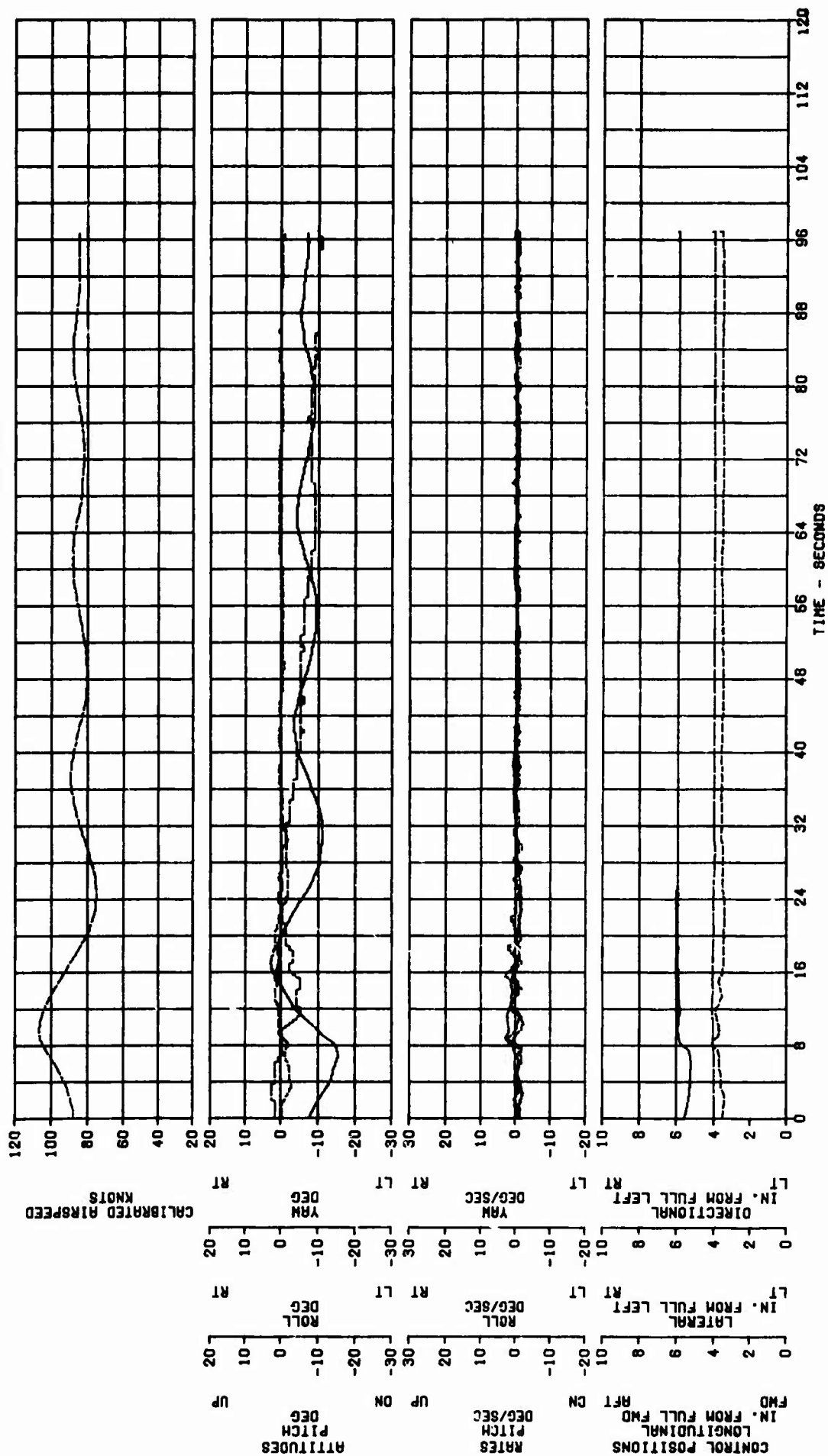


FIGURE 39  
LONGITUDINAL LONG TERM RESPONSE  
JON-68A S/N 68-16708  
MODEL T63-R-700 ENGINE

CROSS METRIC 2920 CG LOCATION 107.7 (FWD) DENSITY ALTITUDE 5860 ORT - DEG C -17.0 ROTOR SPEED - RPM -364 CALIBRATED AIRSPEED - KTS 41 THRUST COEFFICIENT 0.00348 CONFIGURATION CONFIG. C

SOLID LINE SHORT DASH LONG DASH

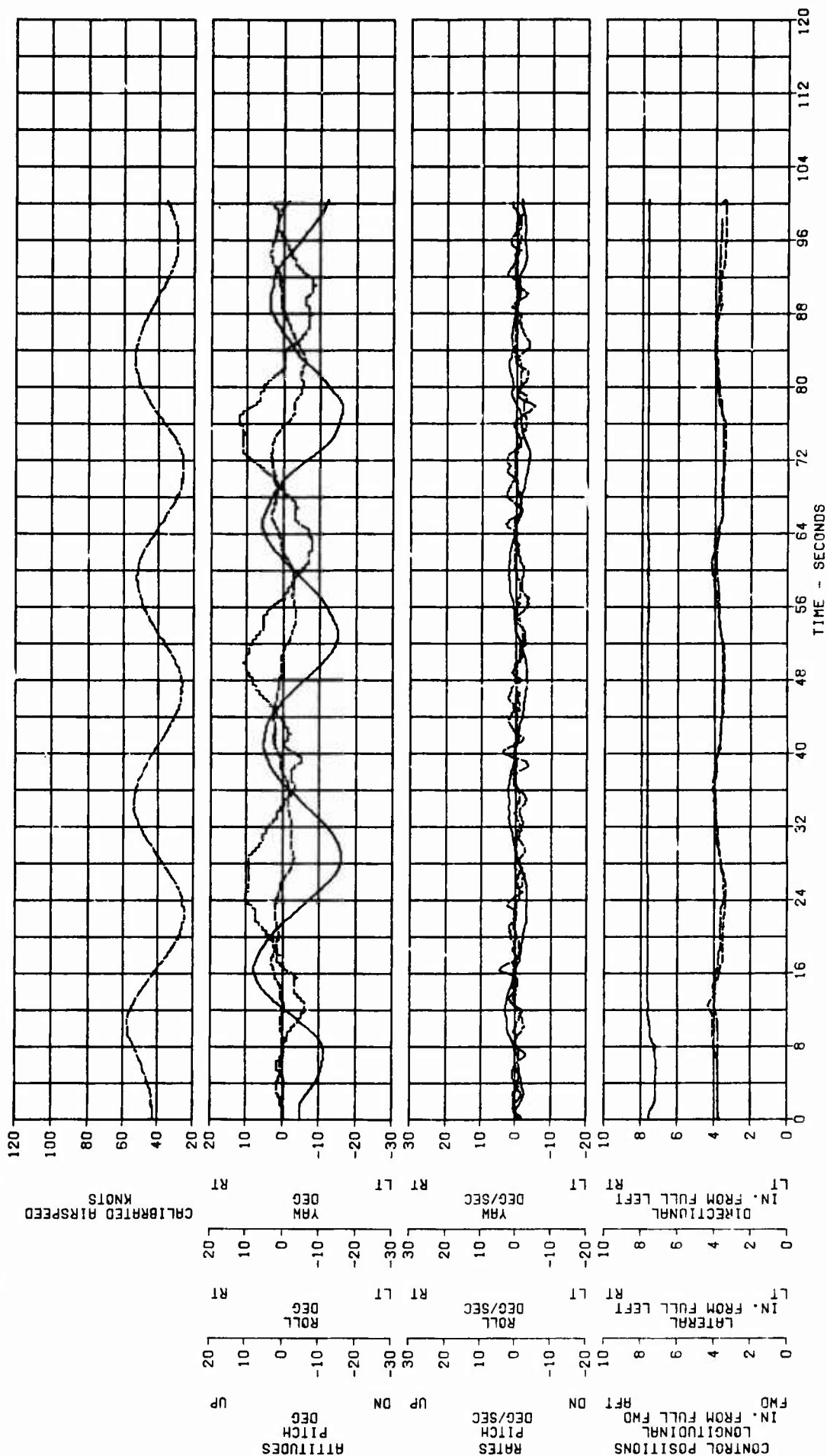




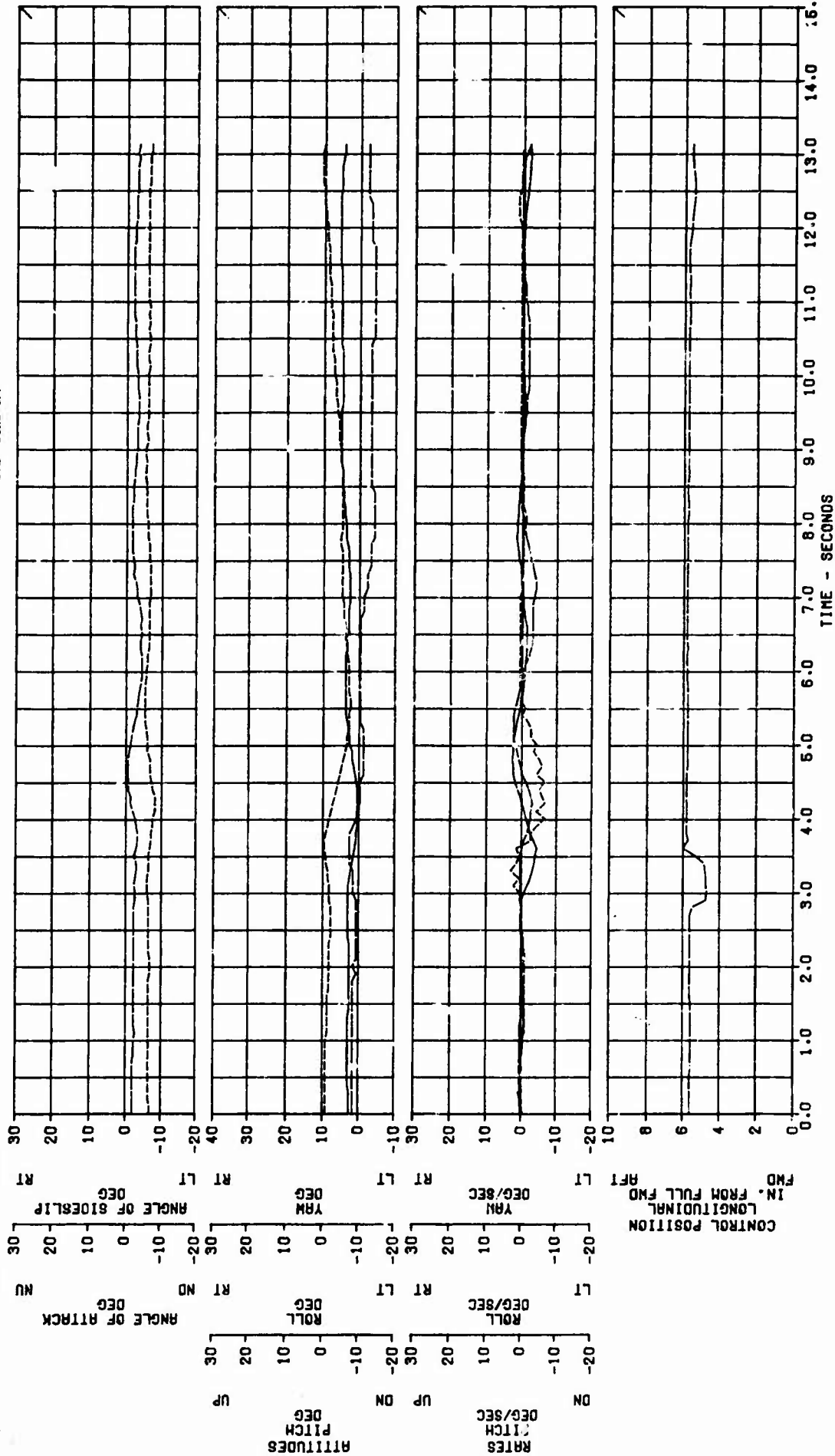


FIGURE 41  
LONGITUDINAL PULSE  
MODEL 100-4-700 220/110

GROSS WEIGHT - LB 2780  
CD LOCATION - IN 107.5 (700)  
DENSITY ALTITUDE - FT 7500  
WAT - DEG C 14.0  
ROTOR SPEED - RPM 364  
CALIBRATED AIRSPEED - KTS 95  
THRUST COEFFICIENT 0.00345  
CONFIGURATION COMFIG. C  
FLIGHT CONDITION LEVEL

NOTE: SC38 OFF

80/10 LINE  
SHORT DRSH  
LONG DRSH



# FIGURE 42 RELEASE FROM STEADY SIDESLIP

JOH-SEA 8/A 73-10708  
MODEL T83-R-30 ENGINE

FLIGHT  
CONDITION  
LEVEL

CONFIGURATION  
CONFIG. A

THRUST  
COEFFICIENT  
0.00362

CALIBRATED  
AIRSPEED  
- KTS 42

ROTOR  
SPEED  
- RPM 364

ORT  
- DEG C 42.6

DENSITY  
ALTITUDE  
- FT 8340

CO  
LOCATION  
- IN 107.6 (FWD)

GROSS  
WEIGHT  
- LB 2740

NOTE: SCRS OFF

LONG  
DASH

SHORT  
DASH

SOLID  
LINE

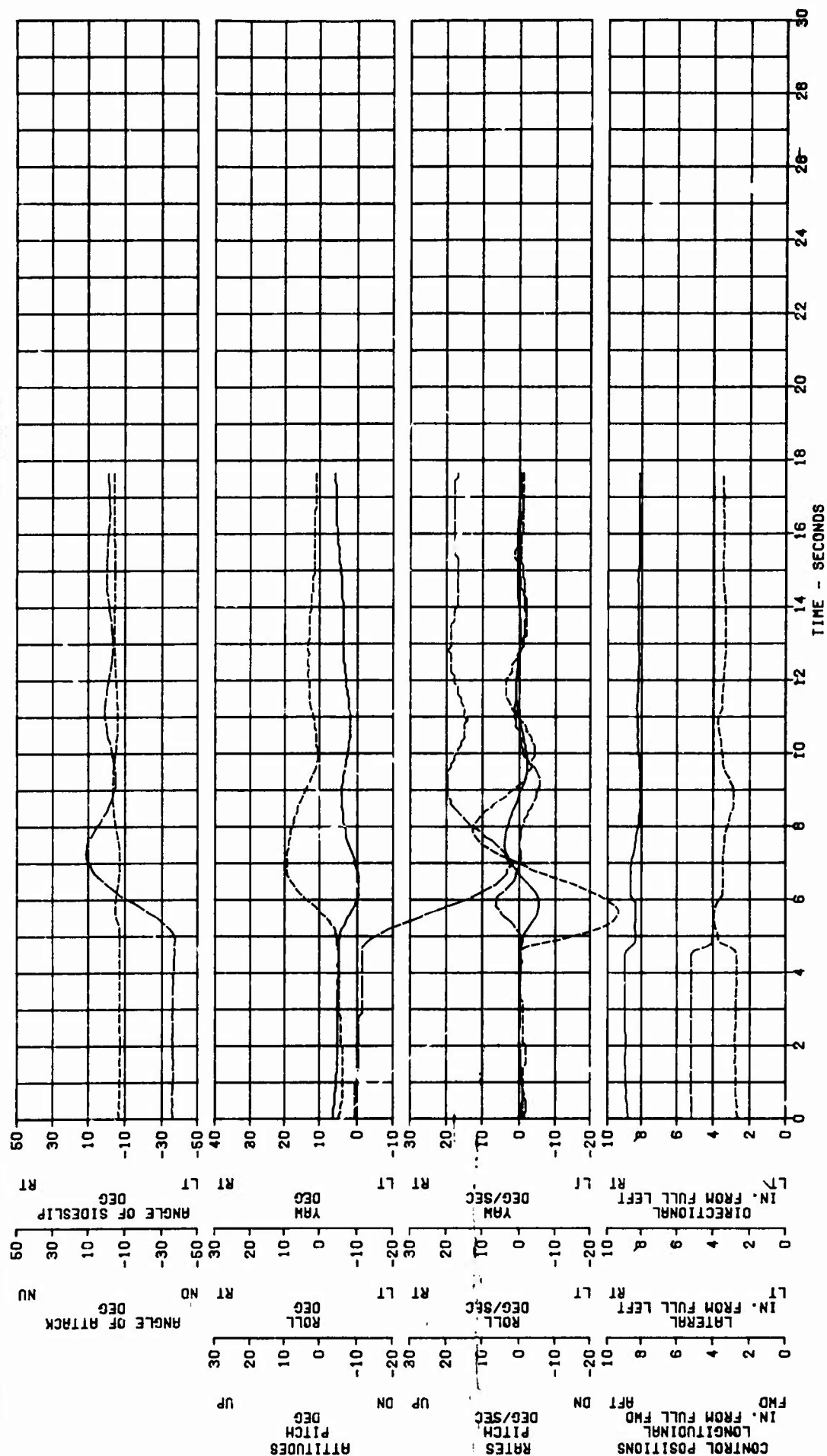


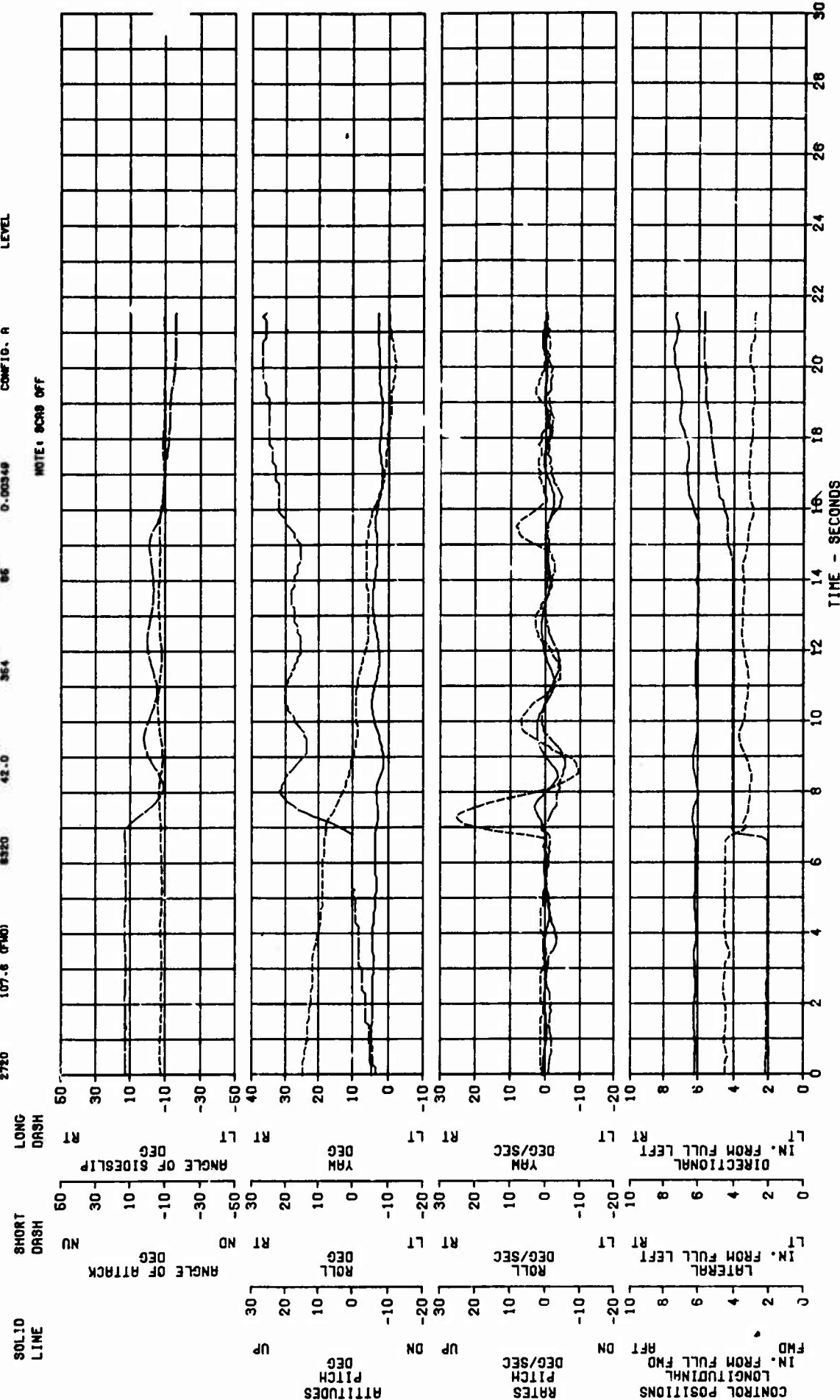
FIGURE 43

RELEASE FROM STEADY SIDESLIP

JOHN-688 9/A 88-16708  
MODEL T83-A-700 ENGINE

WEIGHT  
2780  
CS  
LOCATION  
107.6 (FWD)  
DENSITY  
ALTITUDE  
8320  
DRT  
- DEG C  
42.0  
RPM  
364  
CALIBRATED  
AIRSPEED  
- KTS  
85  
THRUST  
COEFFICIENT  
0.00348  
CONFIGURATION  
CONFIO. A  
FLIGHT  
CONDITION  
LEVEL

NOTE: 8008 OFF





# FIGURE 44 RELEASE FROM STEADY SIDESLIP

JOH-568 3/4 83-1870  
MODEL 183-A-700 ENGINE

FLIGHT  
CONDITION  
LEVEL

CONFIGURATION  
CONFIG. B

THRUST  
COEFFICIENT  
0.00336

CALIBRATED  
AIRSPEED  
- KTS 83

ROTOR  
SPEED  
- RPM 364

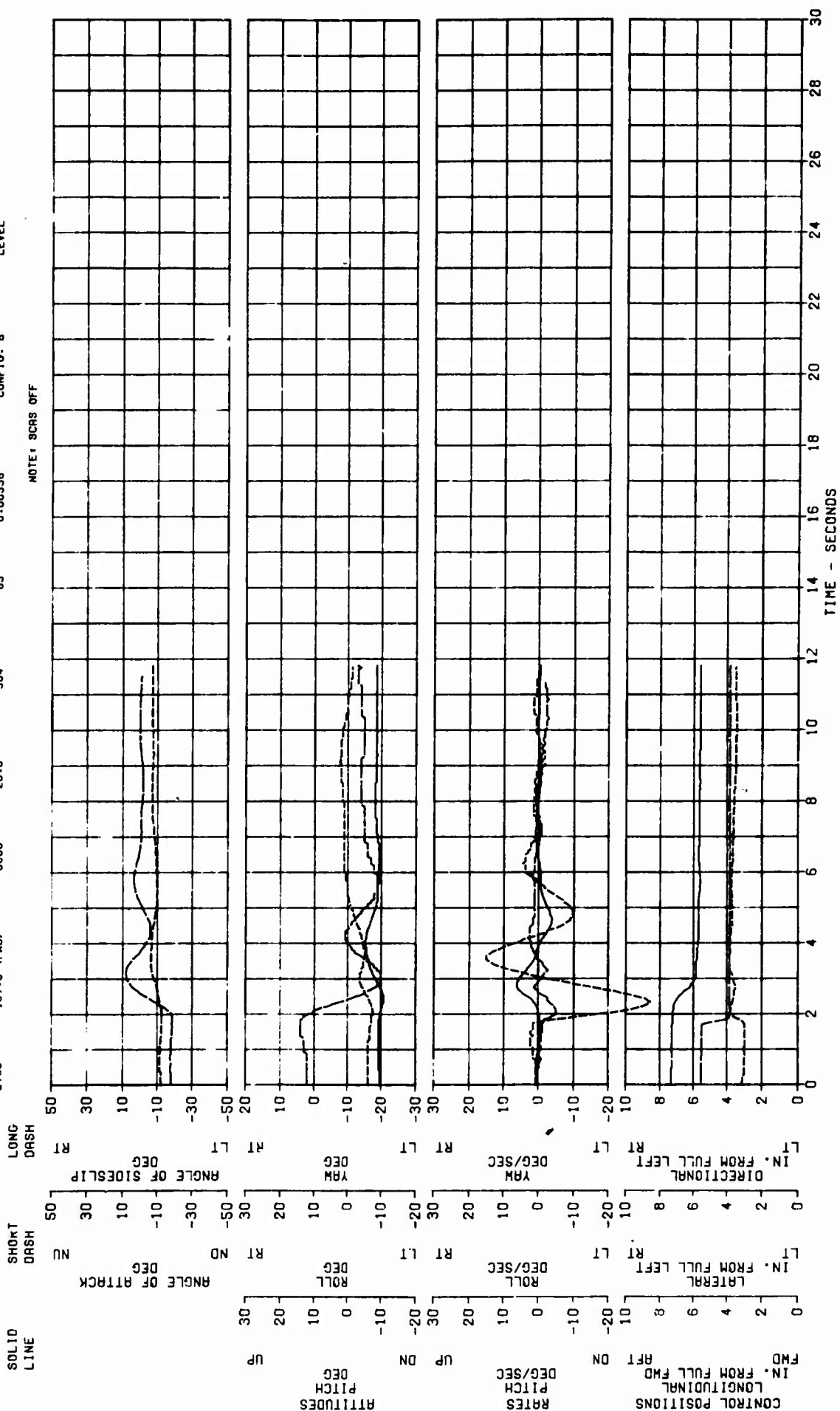
ORT  
- DEC C 20.5

DENSITY  
ALTITUDE  
- FT 6580

CG  
LOCATION  
- IN 107.6 (FWD)

GROSS  
WEIGHT  
- LB 2760

NOTE: SCRS OFF



# RELEASE FROM STEADY SIDESLIP

FIGURE 45  
 RELEASE FROM STEADY SIDESLIP  
 MODEL T33-A-700 ENGINE

FLIGHT  
 CONDITION  
 LEVEL

CONFIGURATION  
 CONF 19. C

THrust  
 COEFFICIENT  
 0.00387

CHLORATED  
 STABILIZER  
 - RT 12

ENGINE  
 SPEED  
 - RPM  
 364

WAT  
 - DEG C  
 14.0

DENSITY  
 ALTITUDE  
 - FT  
 9100

CD  
 LOCATION  
 - IN  
 107.8 (PMO)

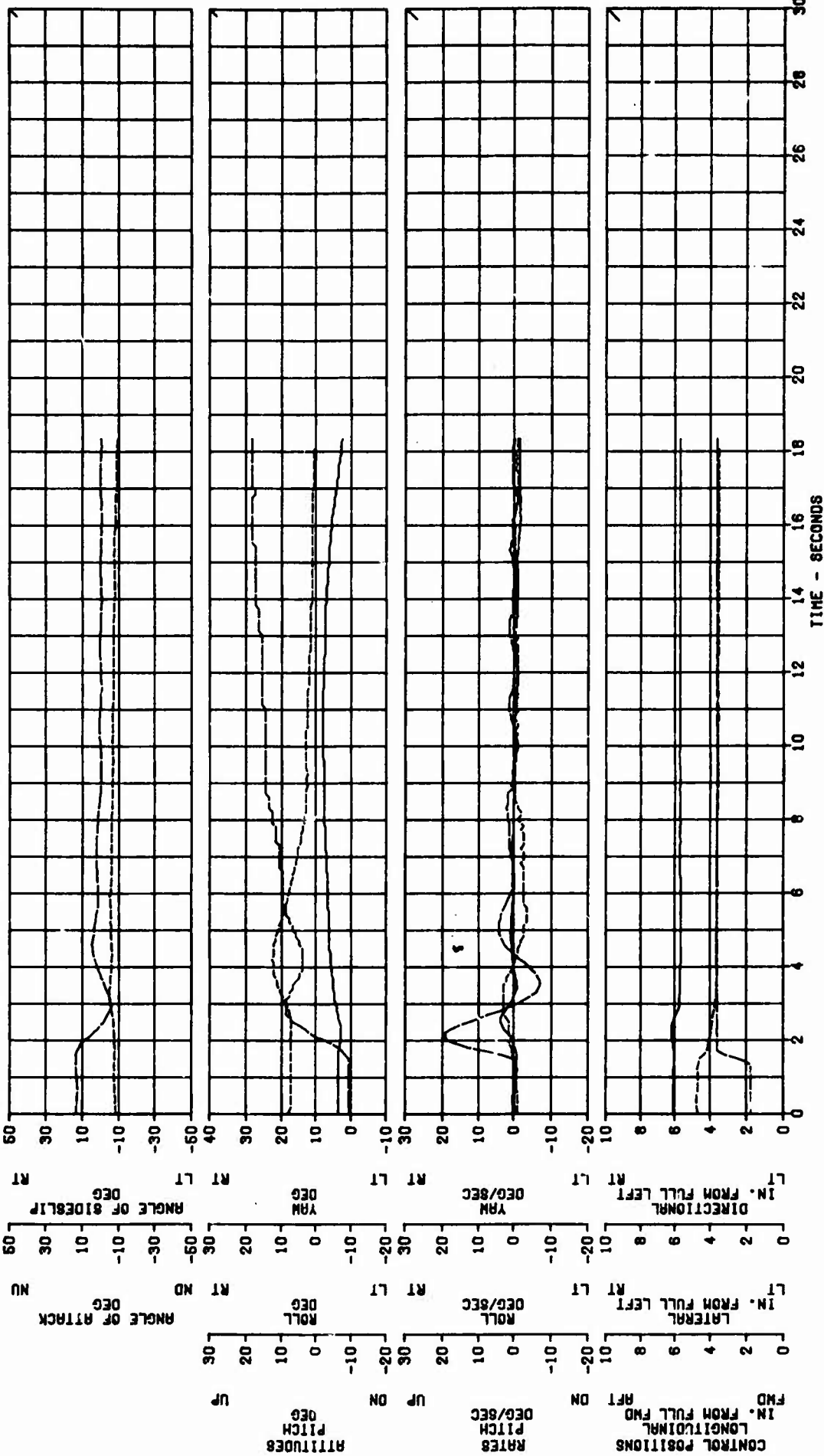
GROSS  
 WEIGHT  
 - LB  
 2800

NOTE: SCHEMATIC

LONG  
 DASH

SHORT  
 DASH

SOLID  
 LINE



# FIGURE 46 RELEASE FROM STEADY SIDESLIP

FLIGHT  
CONDITION  
LEVEL

CONFIGURATION  
CONF10. C

THRUST  
COEFFICIENT  
0.00360

CALIBRATED  
ATTITUDE  
- RTG  
95

ROTOR  
SPEED  
- RPM  
364

OUT  
- DEG C  
14.0

DENSITY  
ALTITUDE  
- FT  
9340

CG  
LOCATION  
- IN  
107.0 (700)

GROSS  
WEIGHT  
- LB  
2000

NOTE: SCRS OFF

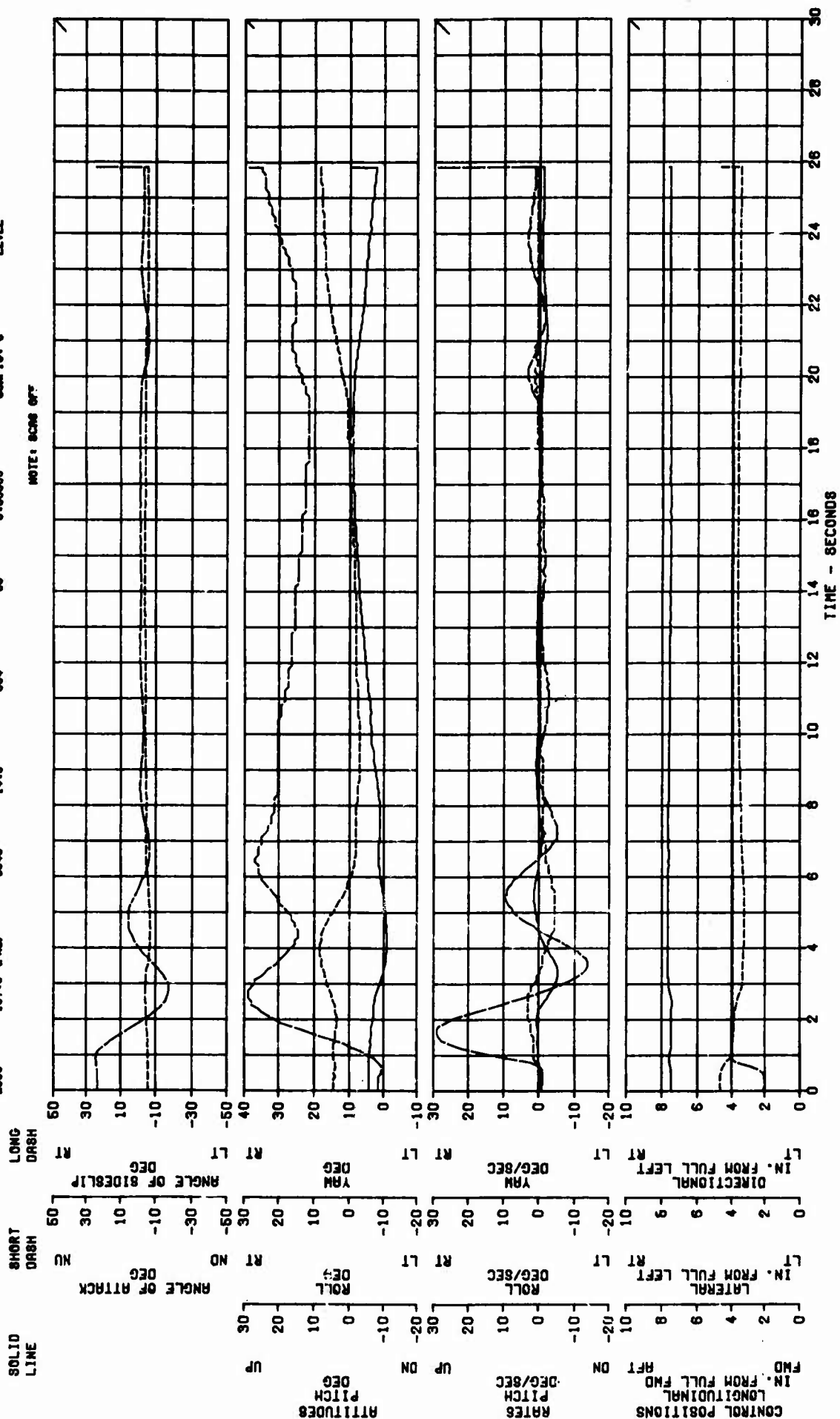


FIGURE 47  
SPIRAL STABILITY  
CONFIG 16. A  
MODEL 163-A-700 ENGINE

FLIGHT  
CONDITION  
LEVEL

CONFIGURATION  
CONF 16. A

THRUST  
COEFFICIENT  
0.00360

CALIBRATED  
AIRSPEED  
- KTS  
87

ROTOR  
SPEED  
- RPM  
364

DAY  
- DEG C  
22.0

DENSITY  
ALTITUDE  
- FT  
8400

CD  
LOCATION  
107.5 IN

CROSS  
WEIGHT  
- LB  
2780

LONG  
DASH

SHORT  
DASH

SOLID  
LINE

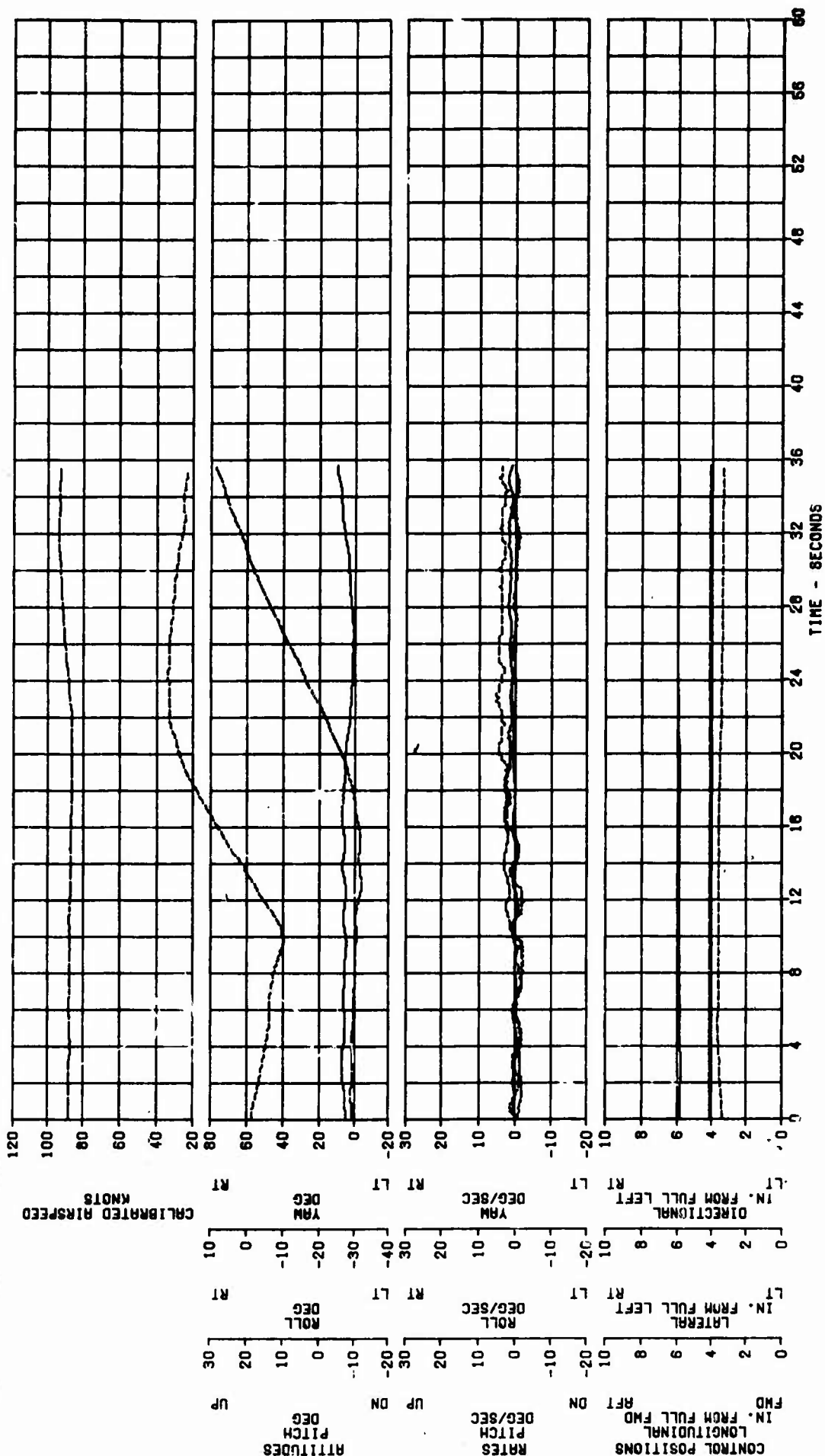




FIGURE 48  
SPIRAL STABILITY  
JOH-588 8/M 88-16708  
MODEL T63-A-700 ENGINE

FLIGHT  
CONDITION  
LEVEL

CONFIGURATION  
CONF19. C

THRUST  
COEFFICIENT  
0.00342

CALIBRATED  
AIRSPEED  
- KTS  
87

ROTOR  
SPEED  
- RPM  
364

GAT  
- DEG C  
18.0

DENSITY  
ALTITUDE  
- FT  
6260

CG  
LOCATION  
- IN  
107.6 (FWD)

GROSS  
WEIGHT  
- LB  
2940

LONG  
DASH

SHORT  
DASH

SOLID  
LINE

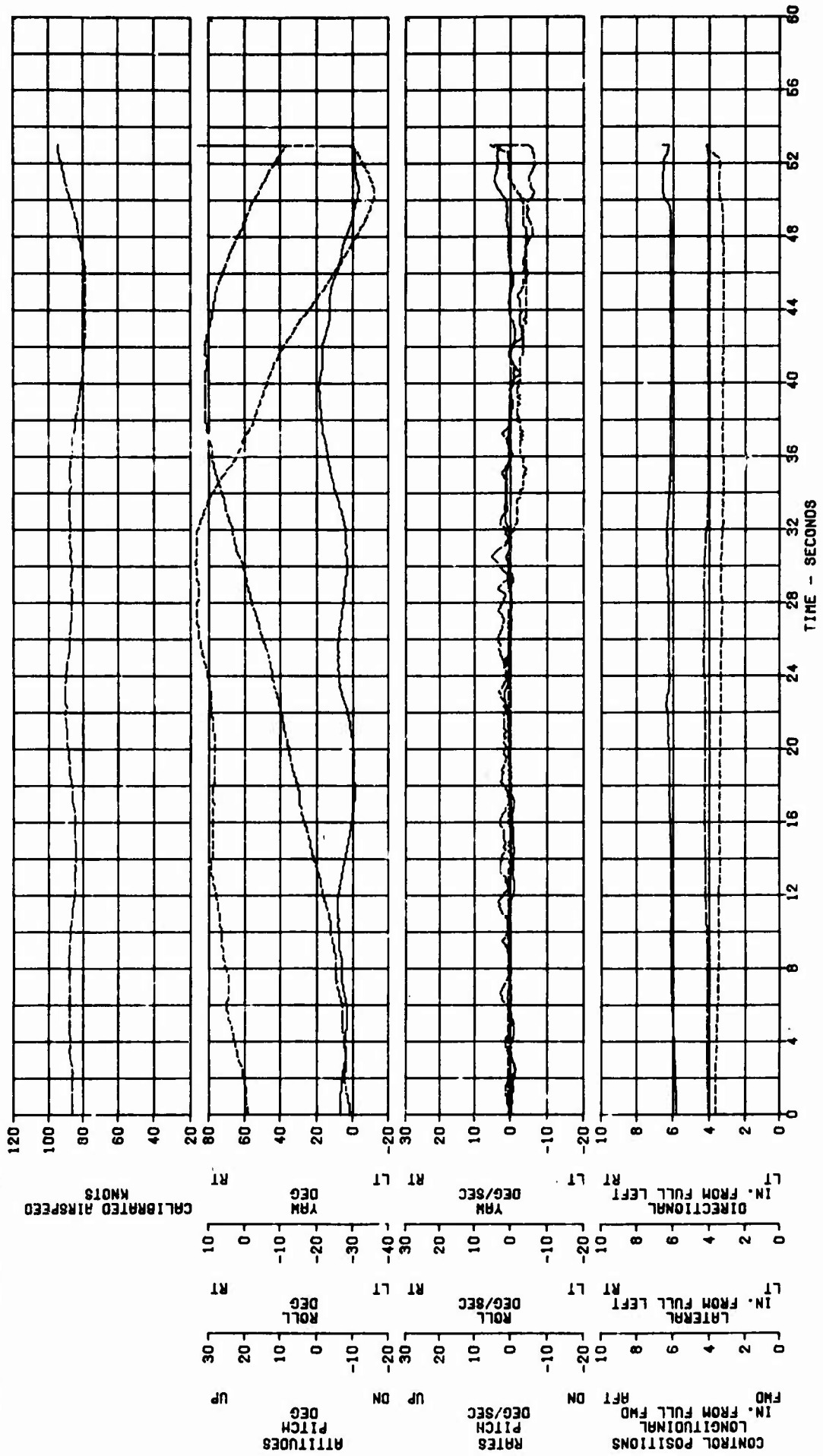


FIGURE 49  
SPIRAL STABILITY  
JANUARY 8, 1964  
MODEL 163-A-700 ENGINE

GROSS WEIGHT - LB 2000  
 CG LOCATION - IN 107.6 (FM0)  
 DENSITY ALTITUDE - FT 7220  
 GWT - DEC C 14.0  
 ROTOR SPEED - RPM 364  
 CALIBRATED AIRSPEED - KTS 87  
 THRUST COEFFICIENT 0.00348  
 CONFIGURATION COMF19. C  
 FLIGHT CONDITION LEVEL

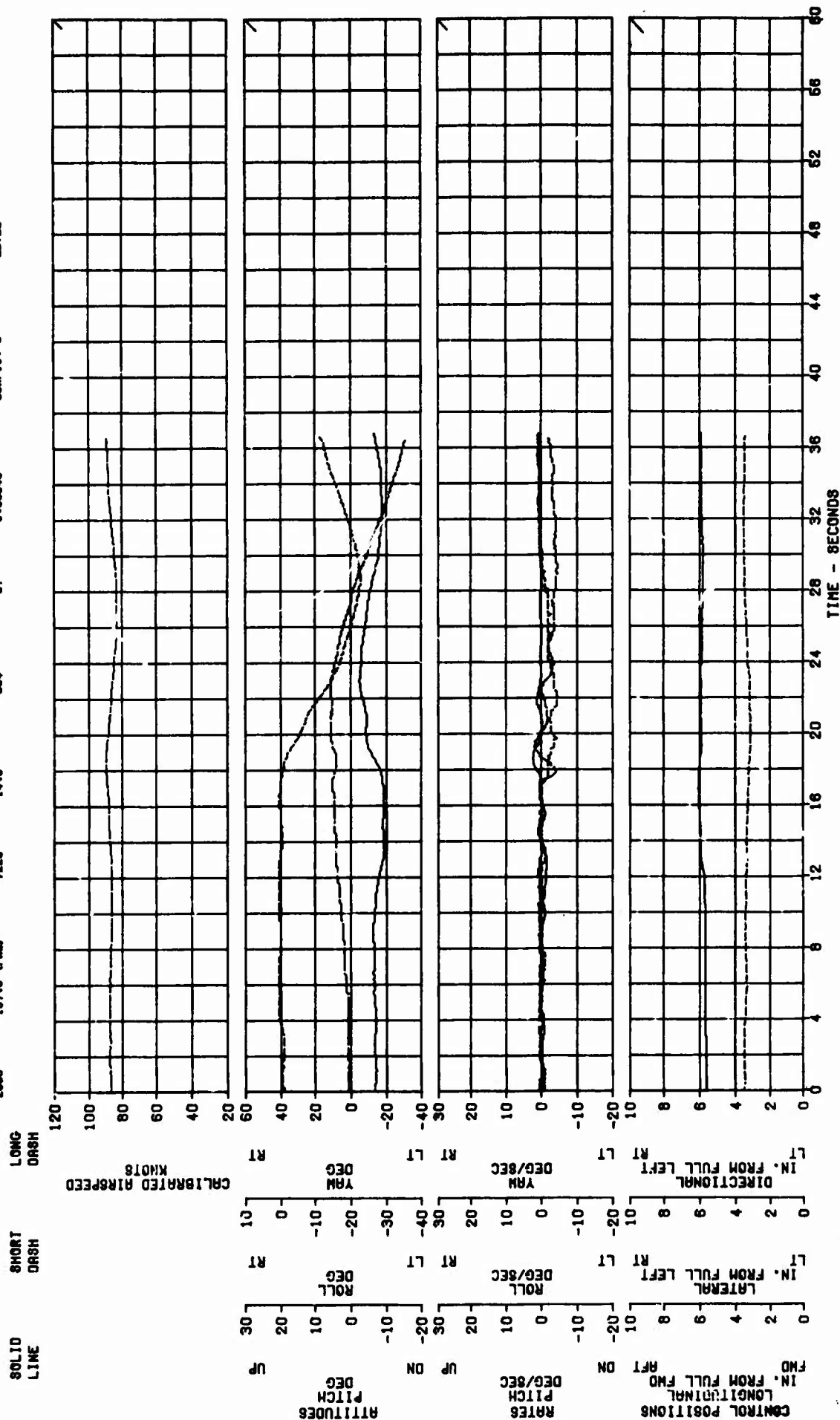


FIGURE 50  
SPIRAL STABILITY  
JAN-68 S/N 88-15708  
MODEL 183-A-700 ENGINE

FLIGHT  
CONDITION  
LEVEL

CONFIGURATION  
CONF10. C

THrust  
COEFFICIENT  
0.00348

C LIMITED  
IN SPEED  
F 3

ROTOR  
SPEED  
- RPM  
364

GWT  
- DEG C  
14.5

DENSITY  
ALTITUDE  
- FT  
8940

CS  
LOCATION  
107.6 (7400)

GROSS  
WEIGHT  
- LB  
2800

LONG  
DASH

SHORT  
DASH

SOLID  
LINE

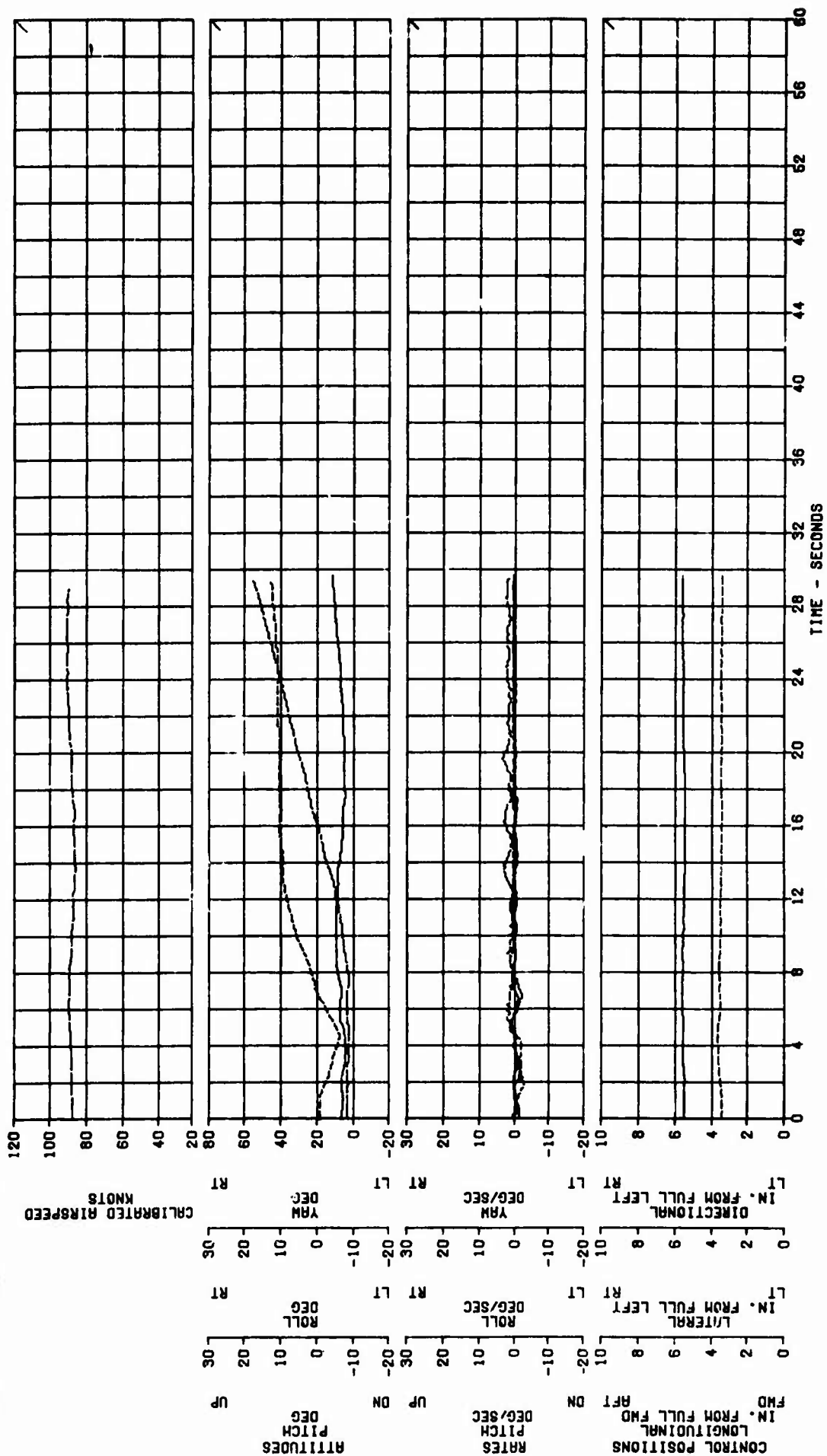


FIGURE 51  
SPIRAL STABILITY  
JAP-688 9/4 68-10708  
MODEL 163-A-700 ENGINE

CROSS  
ALTIMETER  
2000

CG  
LOCATION  
107.6 (FWD)

DENSITY  
ALTITUDE  
6280

QNT  
-DEG C  
-15.0

ROTOR  
SPEED  
RPM  
354

CALIBRATED  
AIRSPEED  
-115  
81

THRUST  
COEFFICIENT  
0.00340

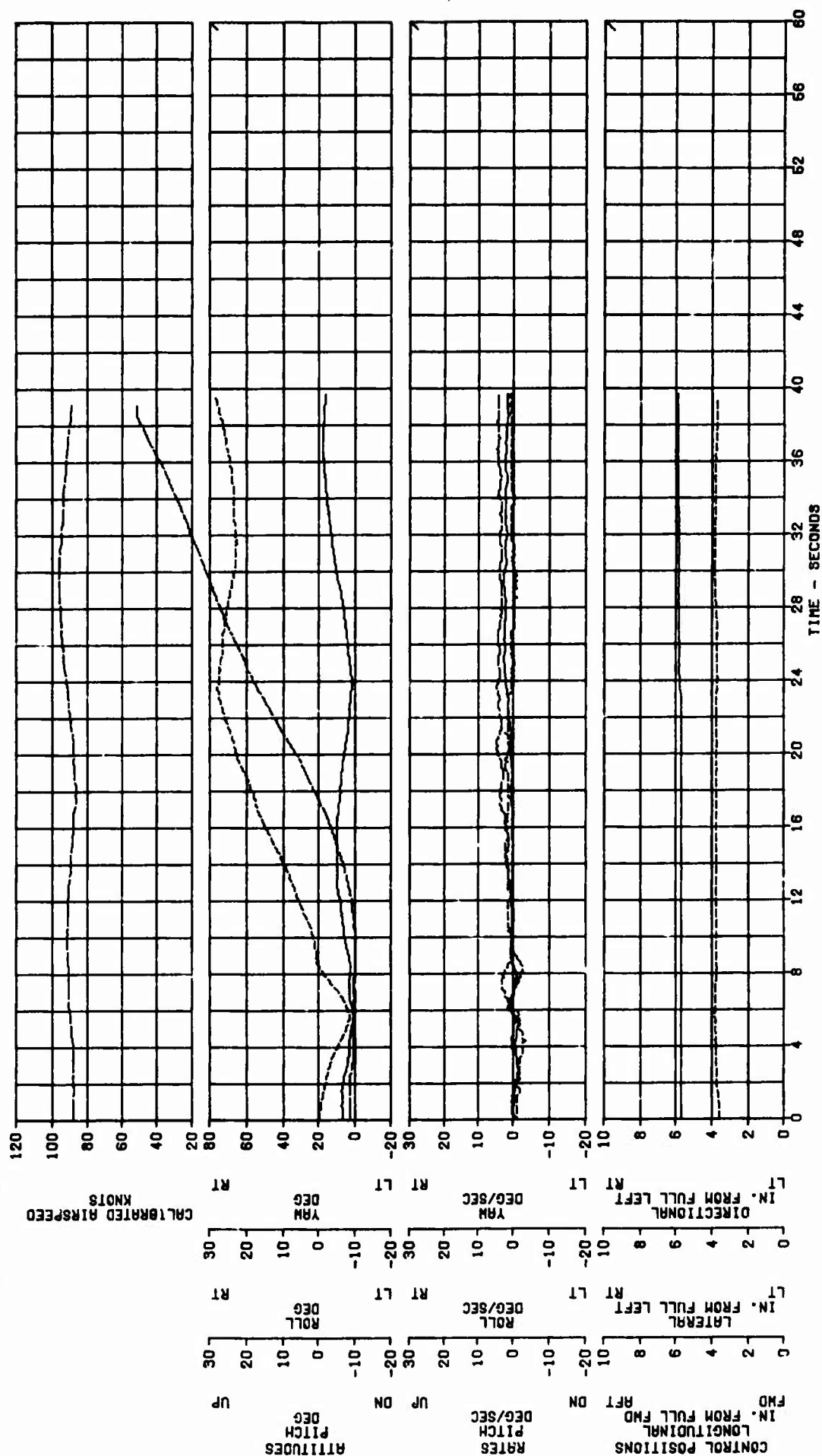
CONFIGURATION  
CONF 10. C

FLIGHT  
CONDITION  
LEVEL

SOLID  
LINE

SHORT  
DASH

LONG  
DASH





### Figure 52 AIRCRAFT REACTION TO SIMULATED SUDDEN ENGINE FAILURE

Figure 52

10 SIMULATED SUN  
JAN-68A 2/1 00-15700  
MODEL 100-A-700 ENGINE

FLIGHT CONDITION	LEVEL
------------------	-------

CONFIDENTIAL

Thrust  
Coefficient

023471  
02160176

2018

out

**QUALITY  
MULTITUDE**

CO  
Location  
- IN  
107-5 07403

**RIGHT**

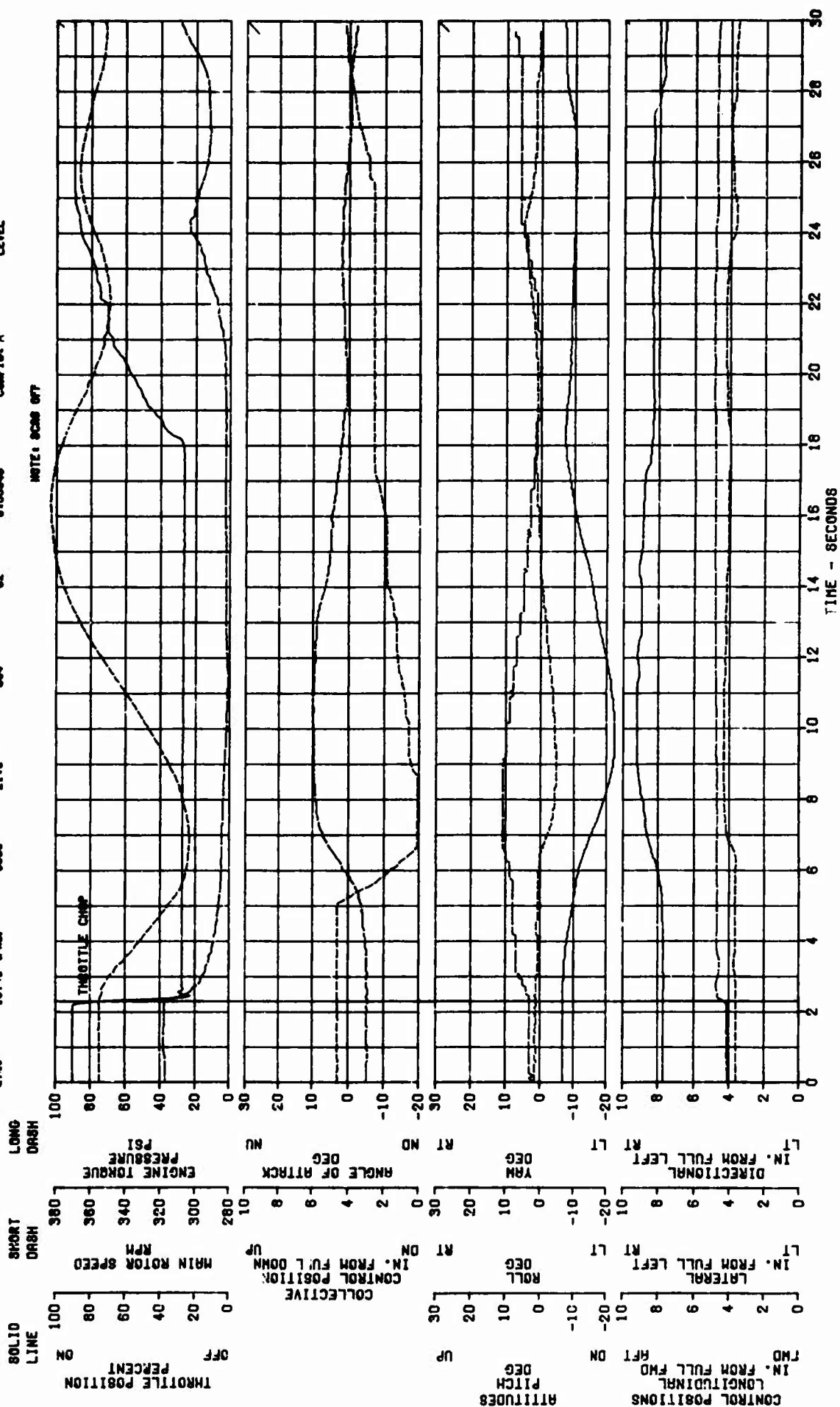


FIGURE 53  
AIRCRAFT REACTION TO SIMULATED SUDDEN ENGINE FAILURE  
JAN-680 8/11 08-18708  
MODEL T63-A-700 ENGINE

ORIGIN  
LOCATION  
107.5 (7400)  
ALTITUDE  
7000  
DENSITY  
- DCD C  
21.0  
ROTOR  
SPEED  
- RPM  
364  
CALCULATED  
RATED  
- KTS  
97  
THRUST  
COEFFICIENT  
0.00346  
CONFIGURATION  
CONF10. A  
FLIGHT  
CONDITION  
LEVEL

NOTE: MCMS OFF

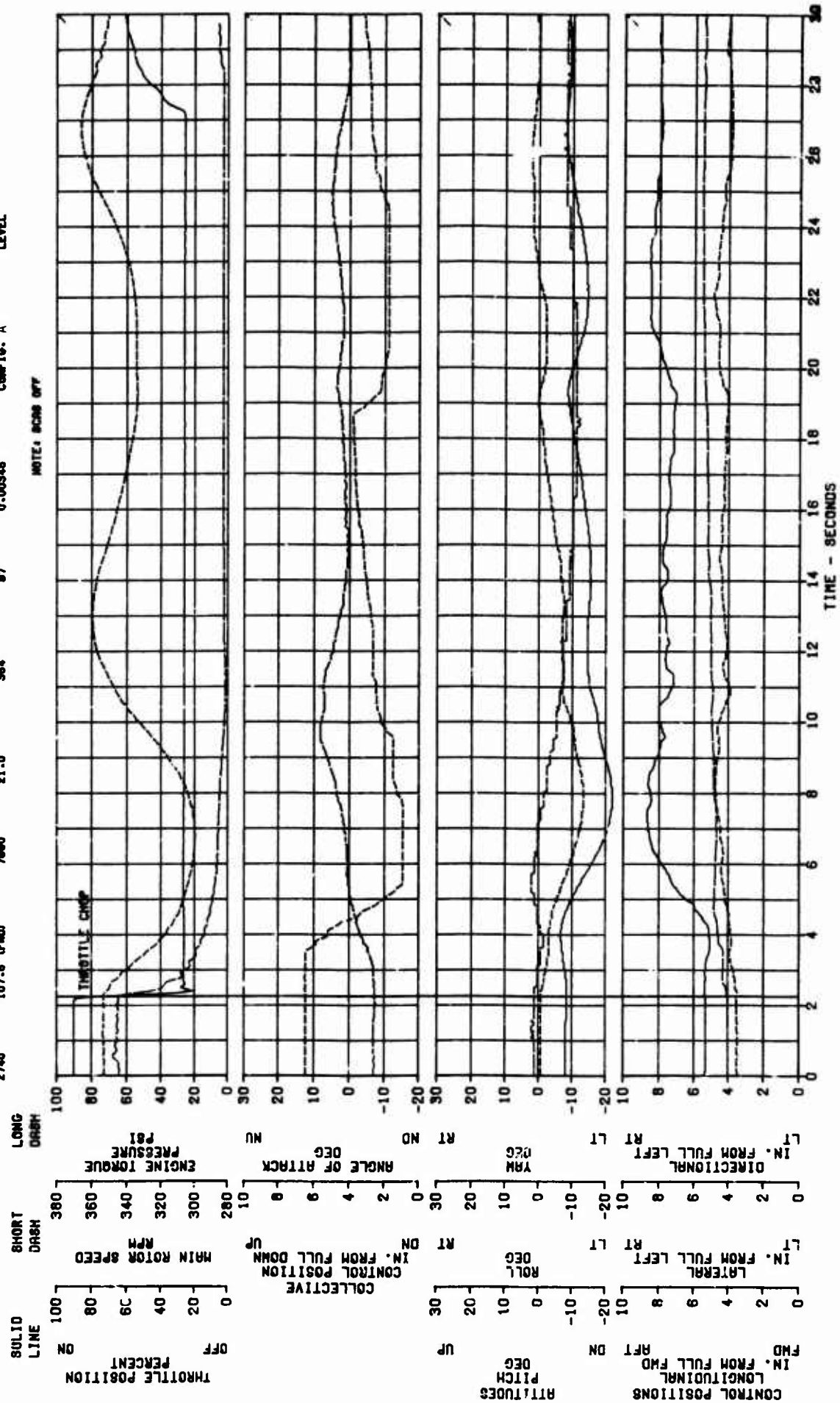


FIGURE 54  
AIRCRAFT REACTION TO SIMULATED SUDDEN ENGINE FAILURE  
JOH-600, S/N 88-18708  
MODEL T63-R-700 ENGINE

FLIGHT  
CONDITION  
LEVEL

CONFIGURATION  
CONF10. C

THRUST  
COEFFICIENT  
0.00342

CALIBRATED  
AIRFUEL  
- 14.0 90

ROTOR  
SPEED  
- 36.1 36.1

GRIT  
- DEG C  
18.0

DENSITY  
ALTITUDE  
5040

CD  
LOCATION  
107.5 (FWD)

GROSS  
WEIGHT  
2880

NOTE: SCRS OFF

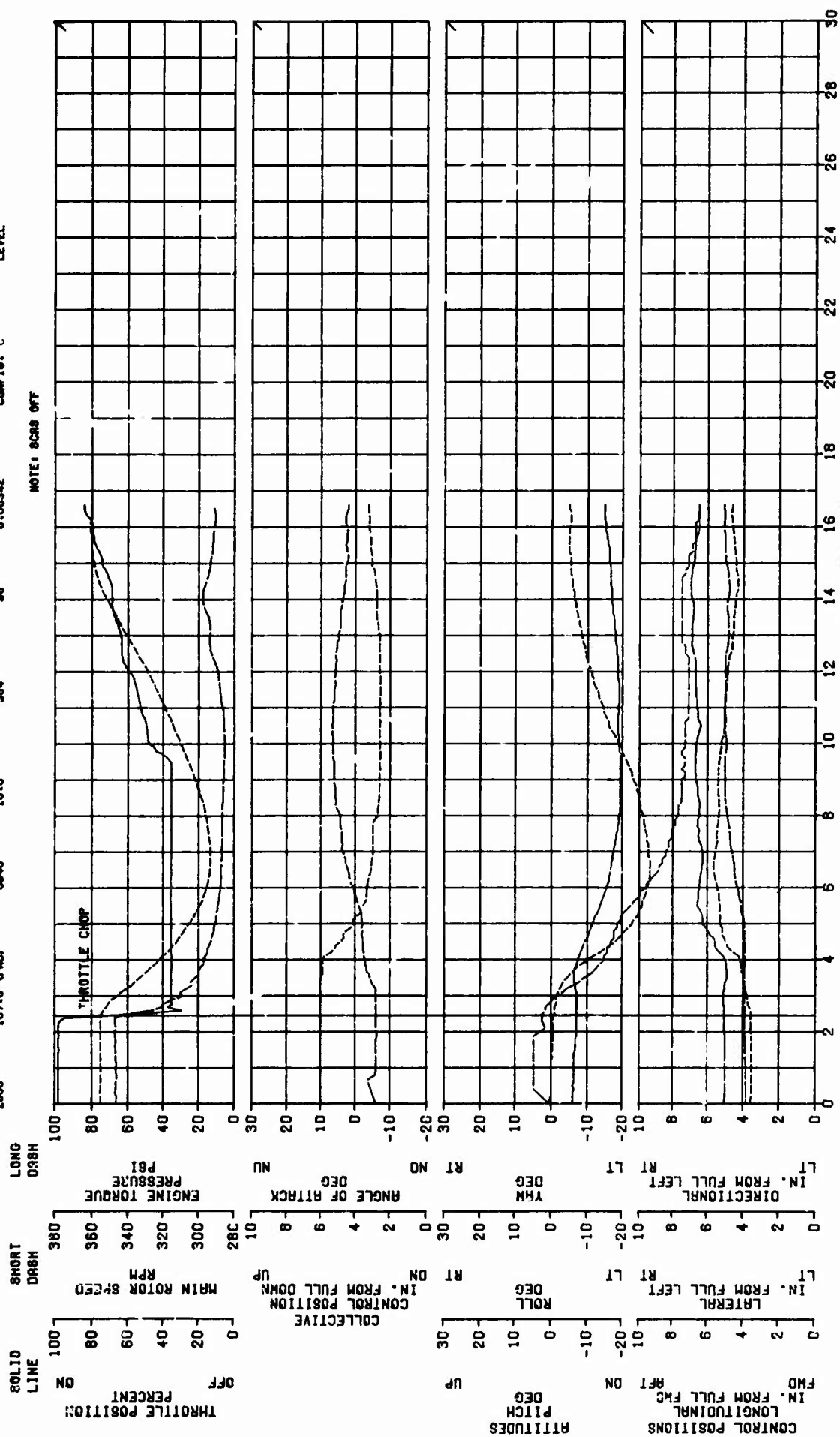


FIGURE 55  
AIRCRAFT REACTION TO SIMULATED SUDDEN ENGINE FAILURE  
CON-OPS 8/11-88-0708  
MODEL T80-A-700 ENGINE

ORIGIN  
WEIGHT  
- LB  
2000

CS  
LOCATION  
107.5 (PMO)

DENSITY  
ALTITUDE  
- FT  
8000

WIND  
- DEG C  
17.0

ENGINE  
APPROX  
- RPM  
364

CALIBRATED  
AIRFUEL  
- KT  
52

THRUST  
COEFFICIENT  
0.00339

CONFIGURATION  
CONF10, C

FLIGHT  
CONDITION  
CLIMB

NOTE: SCAM OFF

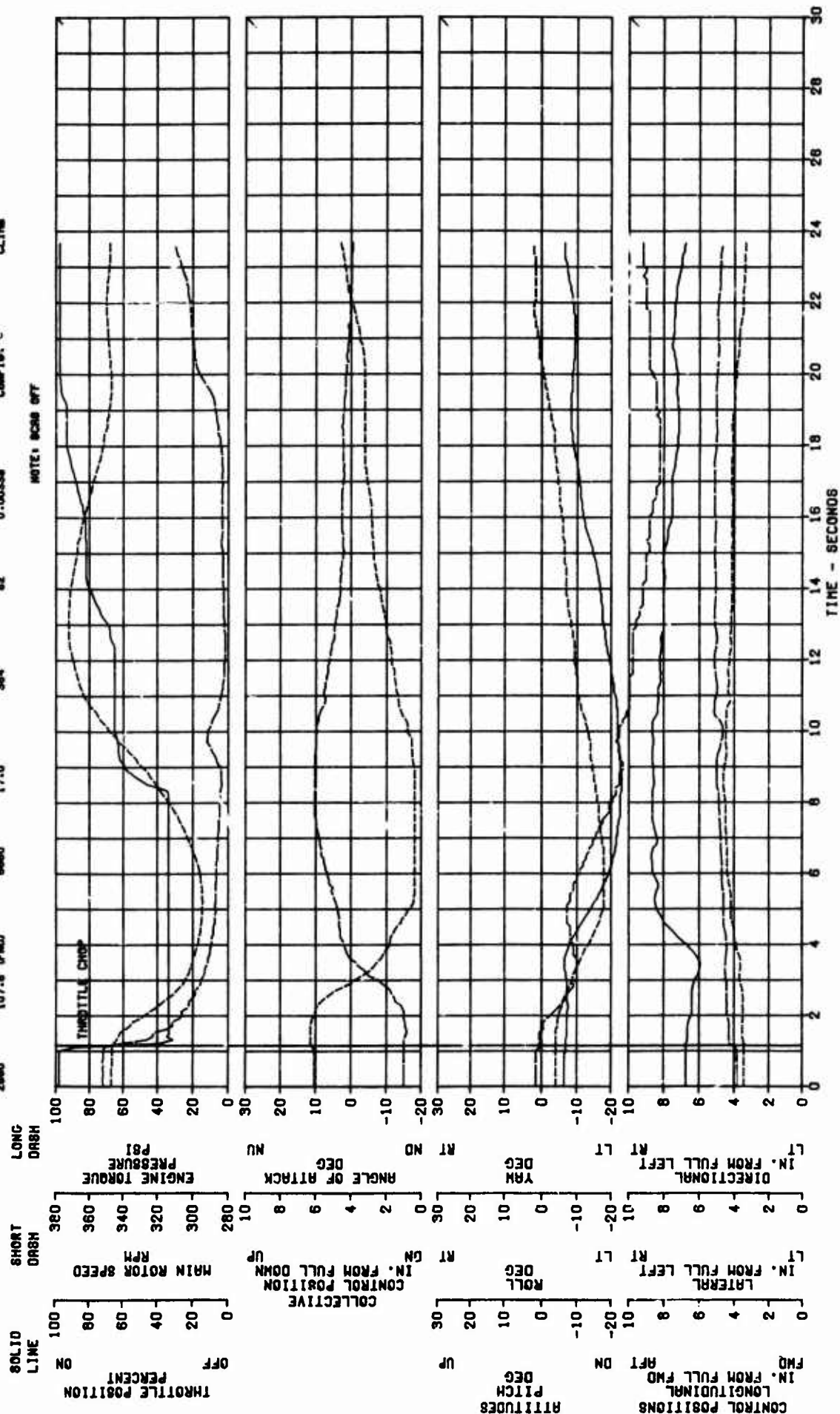
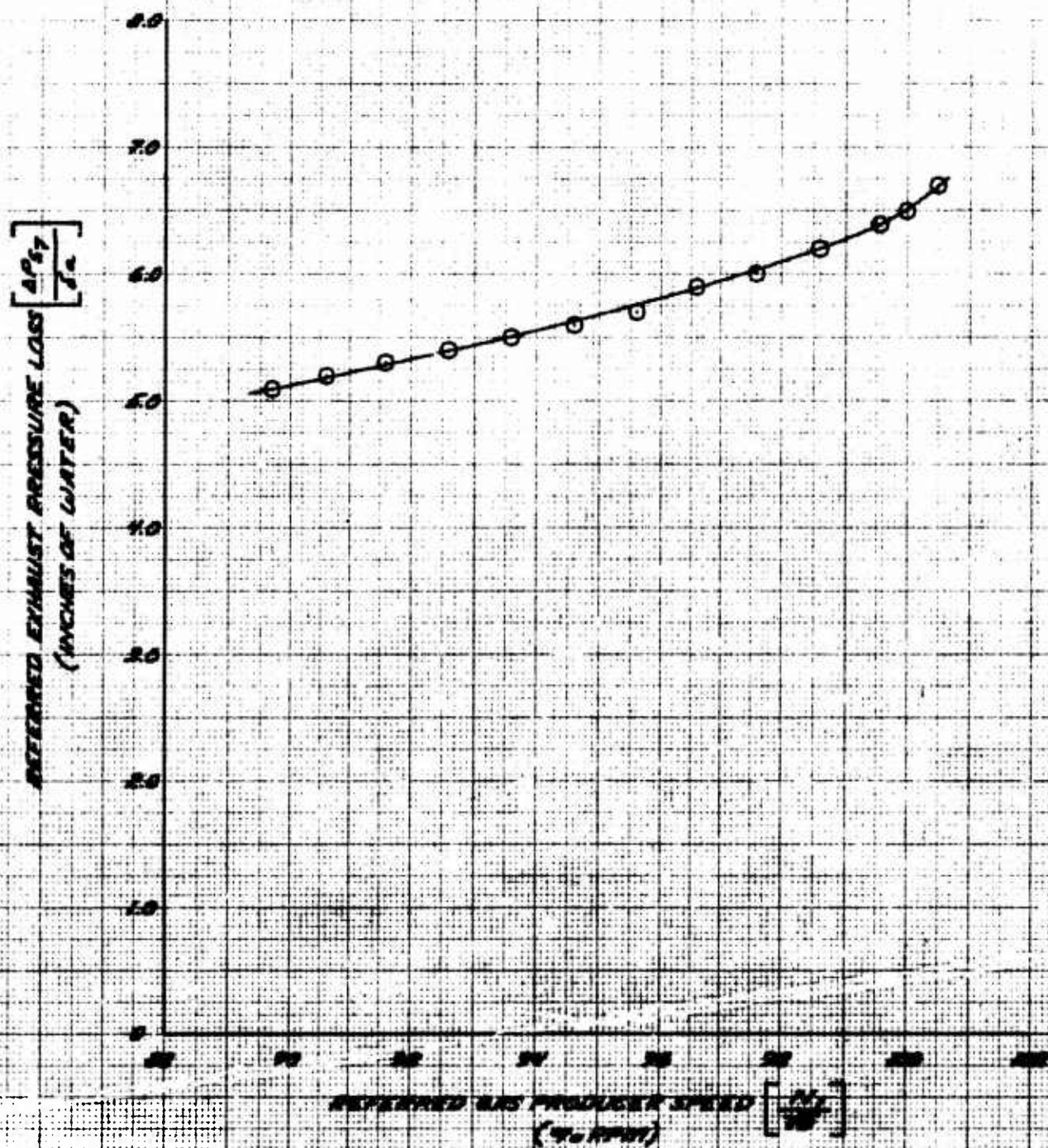




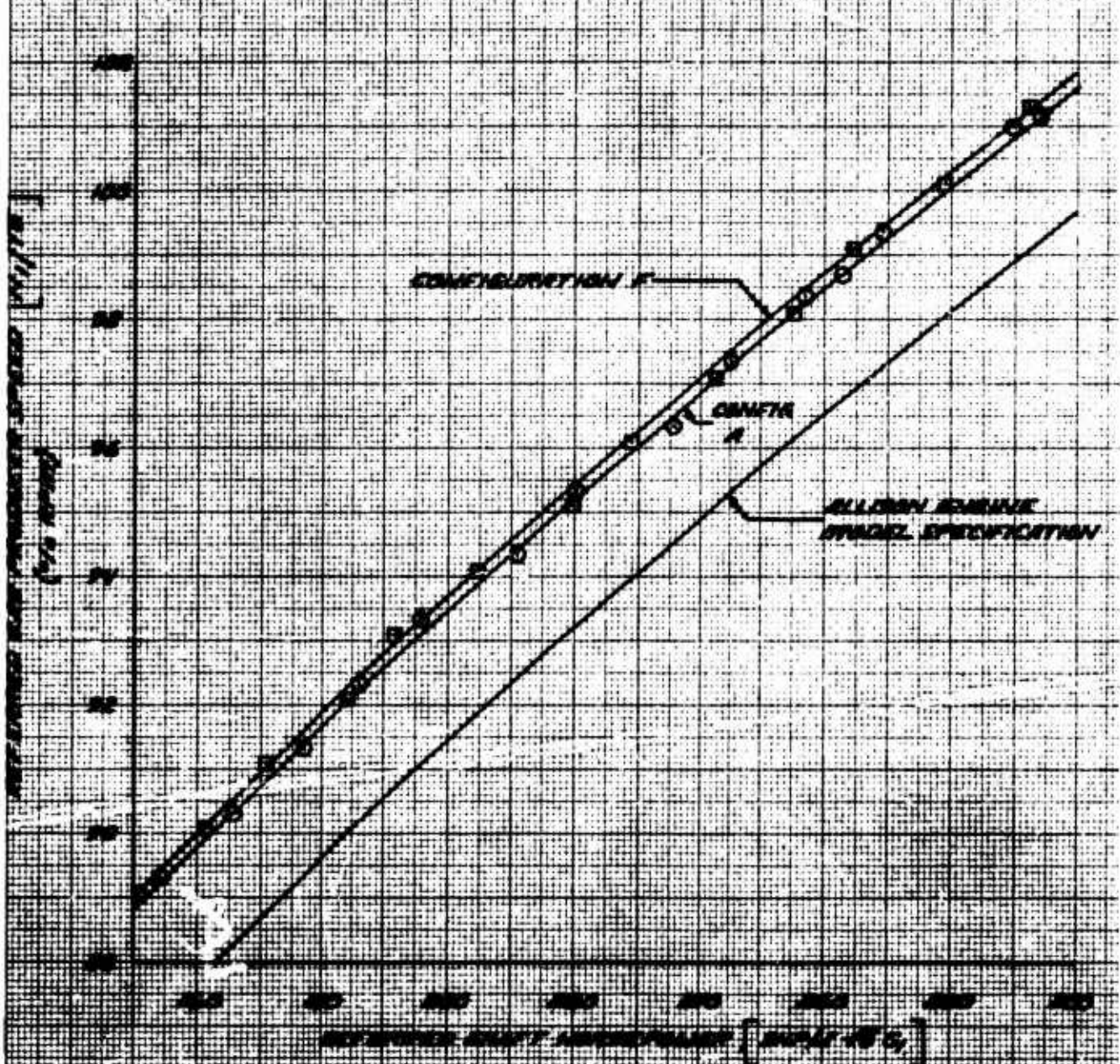
FIGURE 56  
EFFECT OF HANDED EXHAUST EXTENSION  
ON T63-A-700 EXHAUST STATIC PRESSURE  
JOM-58A  
SING68-16706



10-10-50  
 10-10-50  
 10-10-50  
 10-10-50  
 10-10-50

PAGE 12		PAGE 13		PAGE 14		PAGE 15		PAGE 16		PAGE 17		PAGE 18		PAGE 19		PAGE 20		PAGE 21		PAGE 22		PAGE 23		PAGE 24		PAGE 25		PAGE 26		PAGE 27		PAGE 28		PAGE 29		PAGE 30		PAGE 31		PAGE 32		PAGE 33		PAGE 34		PAGE 35		PAGE 36		PAGE 37		PAGE 38		PAGE 39		PAGE 40		PAGE 41		PAGE 42		PAGE 43		PAGE 44		PAGE 45		PAGE 46		PAGE 47		PAGE 48		PAGE 49		PAGE 50		PAGE 51		PAGE 52		PAGE 53		PAGE 54		PAGE 55		PAGE 56		PAGE 57		PAGE 58		PAGE 59		PAGE 60		PAGE 61		PAGE 62		PAGE 63		PAGE 64		PAGE 65		PAGE 66		PAGE 67		PAGE 68		PAGE 69		PAGE 70		PAGE 71		PAGE 72		PAGE 73		PAGE 74		PAGE 75		PAGE 76		PAGE 77		PAGE 78		PAGE 79		PAGE 80		PAGE 81		PAGE 82		PAGE 83		PAGE 84		PAGE 85		PAGE 86		PAGE 87		PAGE 88		PAGE 89		PAGE 90		PAGE 91		PAGE 92		PAGE 93		PAGE 94		PAGE 95		PAGE 96		PAGE 97		PAGE 98		PAGE 99		PAGE 100		PAGE 101		PAGE 102		PAGE 103		PAGE 104		PAGE 105		PAGE 106		PAGE 107		PAGE 108		PAGE 109		PAGE 110		PAGE 111		PAGE 112		PAGE 113		PAGE 114		PAGE 115		PAGE 116		PAGE 117		PAGE 118		PAGE 119		PAGE 120		PAGE 121		PAGE 122		PAGE 123		PAGE 124		PAGE 125		PAGE 126		PAGE 127		PAGE 128		PAGE 129		PAGE 130		PAGE 131		PAGE 132		PAGE 133		PAGE 134		PAGE 135		PAGE 136		PAGE 137		PAGE 138		PAGE 139		PAGE 140		PAGE 141		PAGE 142		PAGE 143		PAGE 144		PAGE 145		PAGE 146		PAGE 147		PAGE 148		PAGE 149		PAGE 150		PAGE 151		PAGE 152		PAGE 153		PAGE 154		PAGE 155		PAGE 156		PAGE 157		PAGE 158		PAGE 159		PAGE 160		PAGE 161		PAGE 162		PAGE 163		PAGE 164		PAGE 165		PAGE 166		PAGE 167		PAGE 168		PAGE 169		PAGE 170		PAGE 171		PAGE 172		PAGE 173		PAGE 174		PAGE 175		PAGE 176		PAGE 177		PAGE 178		PAGE 179		PAGE 180		PAGE 181		PAGE 182		PAGE 183		PAGE 184		PAGE 185		PAGE 186		PAGE 187		PAGE 188		PAGE 189		PAGE 190		PAGE 191		PAGE 192		PAGE 193		PAGE 194		PAGE 195		PAGE 196		PAGE 197		PAGE 198		PAGE 199		PAGE 200		PAGE 201		PAGE 202		PAGE 203		PAGE 204		PAGE 205		PAGE 206		PAGE 207		PAGE 208		PAGE 209		PAGE 210		PAGE 211		PAGE 212		PAGE 213		PAGE 214		PAGE 215		PAGE 216		PAGE 217		PAGE 218		PAGE 219		PAGE 220		PAGE 221		PAGE 222		PAGE 223		PAGE 224		PAGE 225		PAGE 226		PAGE 227		PAGE 228		PAGE 229		PAGE 230		PAGE 231		PAGE 232		PAGE 233		PAGE 234		PAGE 235		PAGE 236		PAGE 237		PAGE 238		PAGE 239		PAGE 240		PAGE 241		PAGE 242		PAGE 243		PAGE 244		PAGE 245		PAGE 246		PAGE 247		PAGE 248		PAGE 249		PAGE 250		PAGE 251		PAGE 252		PAGE 253		PAGE 254		PAGE 255		PAGE 256		PAGE 257		PAGE 258		PAGE 259		PAGE 260		PAGE 261		PAGE 262		PAGE 263		PAGE 264		PAGE 265		PAGE 266		PAGE 267		PAGE 268		PAGE 269		PAGE 270		PAGE 271		PAGE 272		PAGE 273		PAGE 274		PAGE 275		PAGE 276		PAGE 277		PAGE 278		PAGE 279		PAGE 280		PAGE 281		PAGE 282		PAGE 283		PAGE 284		PAGE 285		PAGE 286		PAGE 287		PAGE 288		PAGE 289		PAGE 290		PAGE 291		PAGE 292		PAGE 293		PAGE 294		PAGE 295		PAGE 296		PAGE 297		PAGE 298		PAGE 299		PAGE 300		PAGE 301		PAGE 302		PAGE 303		PAGE 304		PAGE 305		PAGE 306		PAGE 307		PAGE 308		PAGE 309	
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**BELL EXHAUST EXTENSION (CONFIGURATION A)**  
**HUNNEY EXHAUST EXTENSION (CONFIGURATION F)**

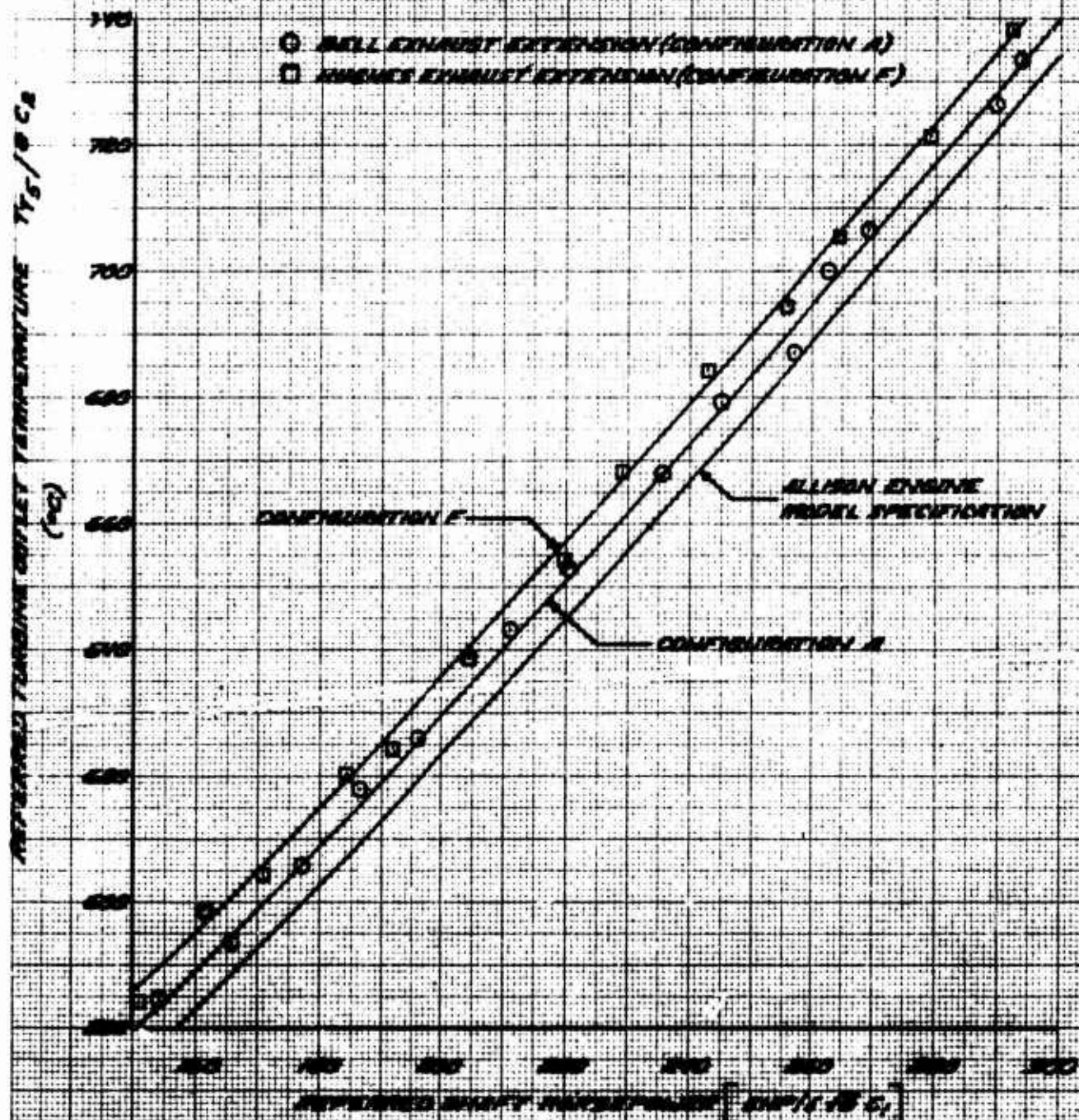




**Figure 23**  
**ENGINE CHARACTERISTICS**  
**300-500**  
**514-524**  
**703-700 51402744**

- NOTE 1** SHIP HULLFAIR CORRECTION FACTOR ( $K_1$ ) AND  $T_{12}$  CORRECTION FACTOR ( $K_2$ ) OBTAINED FROM ALLISON HULL SPECIFICATION FOR SHIP.
- 2**  $T_1$ ,  $P$ ,  $R$ ,  $Q$ , AND  $C_2$  BASED ON COMPRESSOR INLET TOTAL PRESSURE AND TEMPERATURE.
- 3** REED AND 1/2 IN. AND 1/4 IN. ICE OFF.
- 4** POWER EXTRACTED EQUALS 8.0 SHP

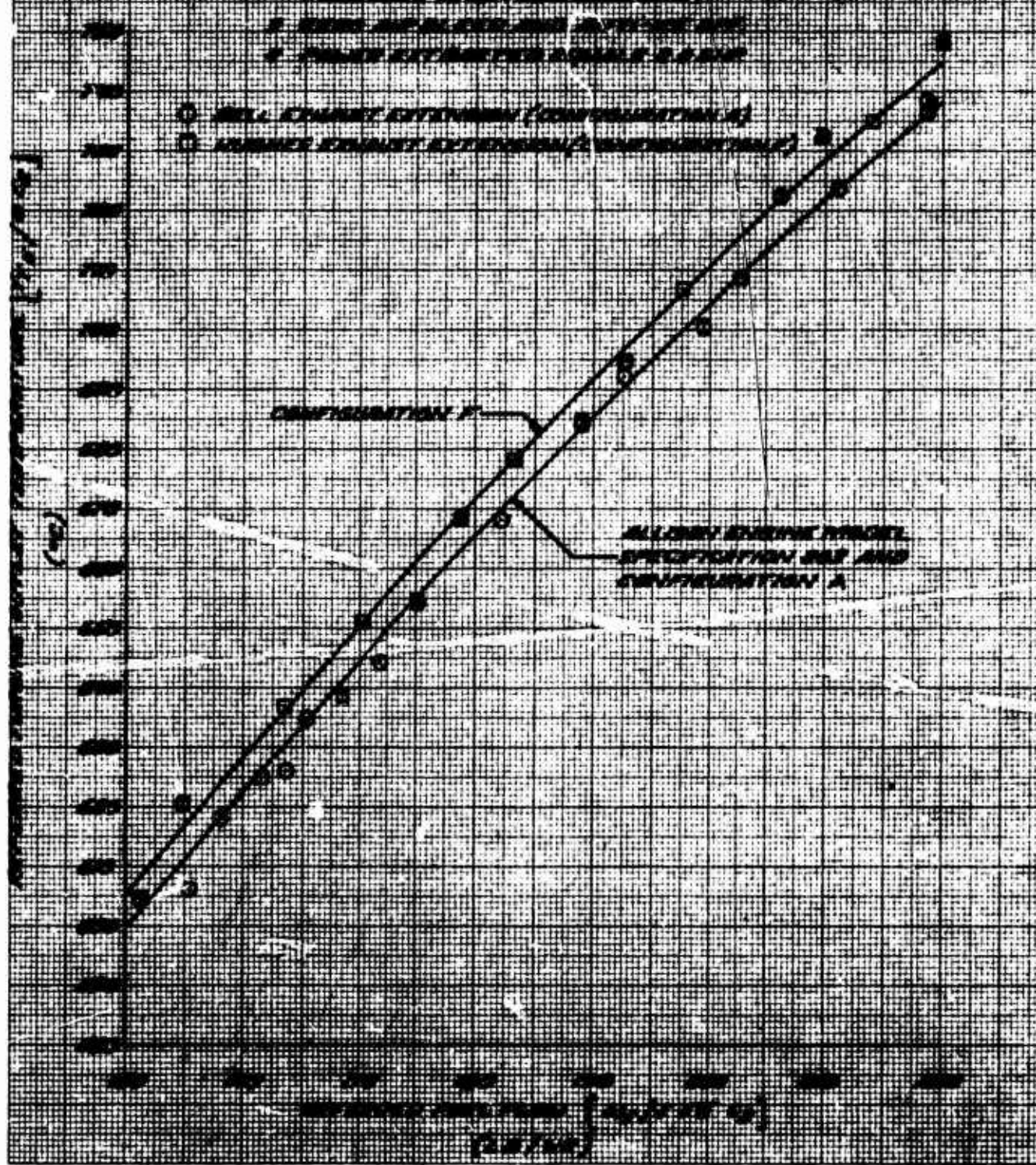
- BELL EXHAUST EXTENSION (CONFIGURATION A)  
 □ ENGINE EXHAUST EXTENSION (CONFIGURATION F)



# Figure 10 Comparison of Predicted and Measured Thrust Extension

1. Thrust extension is plotted against engine power for the two configurations and for the two engine types.
2. The predicted thrust extension is shown by the solid line and the measured thrust extension by the dashed line.
3. The predicted thrust extension is based on the engine power and the engine efficiency.
4. The measured thrust extension is based on the engine power and the engine efficiency.

- MEASURED THRUST EXTENSION (CONFIGURATION A)
- MEASURED THRUST EXTENSION (CONFIGURATION B)



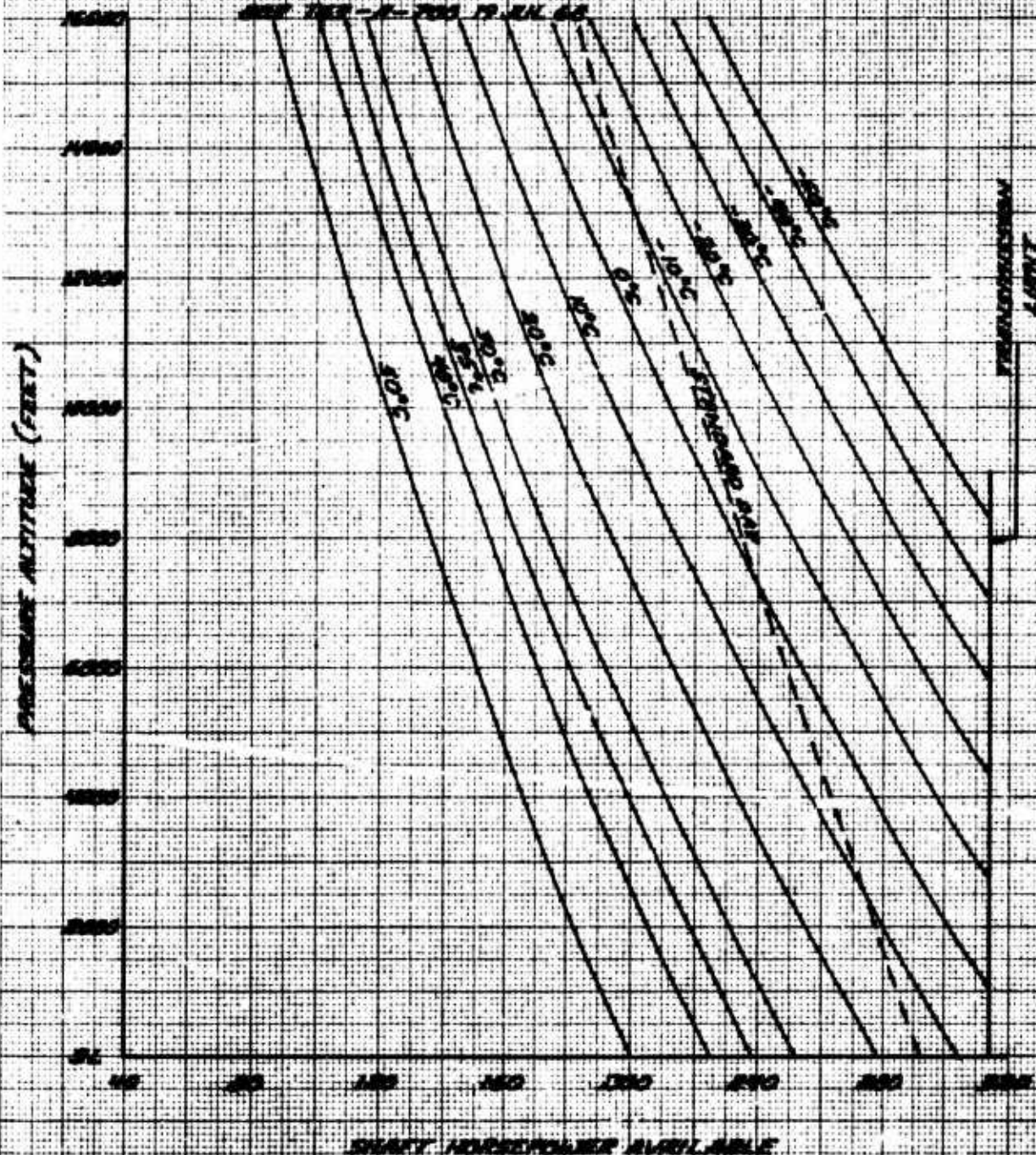


**Figure 60**  
**Start Horsepower Available**  
**JON-504**  
**SIN-66-10706**  
**T63-A-700**

THROTTLE SPEED RANGE 1-2, 1700-2000  
 HUBBARD EXHAUST EXTENSION INSTALLED

**NOTE 1: STATIC CONDITIONS**

1. G VALVE AND SHUT-OFF
2. POWER EXTRACTOR SHALL BE OFF
3. FUEL LEAKAGE BASED ON FIGURES 59 AND 60, CURRENT FINAL REPORT 65-20
4. EXHAUST PRESSURE LOSS BASED ON FIGURES 59 THROUGH 60
5. START HORSEPOWER DERIVED FROM ENGINE MODEL SPECIFICATION SEE T63-A-700 FOR ALL 65



REPRESENTATIVE AND FBI-A-RMS RECALL OF





# Figure 40 Fuel Flow vs. Altitude

ENGINE: TURBO-PROP-4000-1A (1400 HP)

NOTE: 1. FUEL FLOW AND MAX. ENGINE RPM VARY WITH ALTITUDE. MAX. RPM IS 2400. FUEL FLOW IS 1.0 GPM. AT 2400 RPM.

2. STATIC ALTITUDE.

3. 2 BLEED AND ANTI-ICE OFF.

4. POWER EXTRACTED ENGINE 2.0 SHP.

5. INLET LOSS BASED ON FIGURE 59 AND 60 USARMC FINAL REPORT 42-30.

6. EXHAUST PRESSURE LOSS BASED ON FIGURE 57 THROUGH 60.

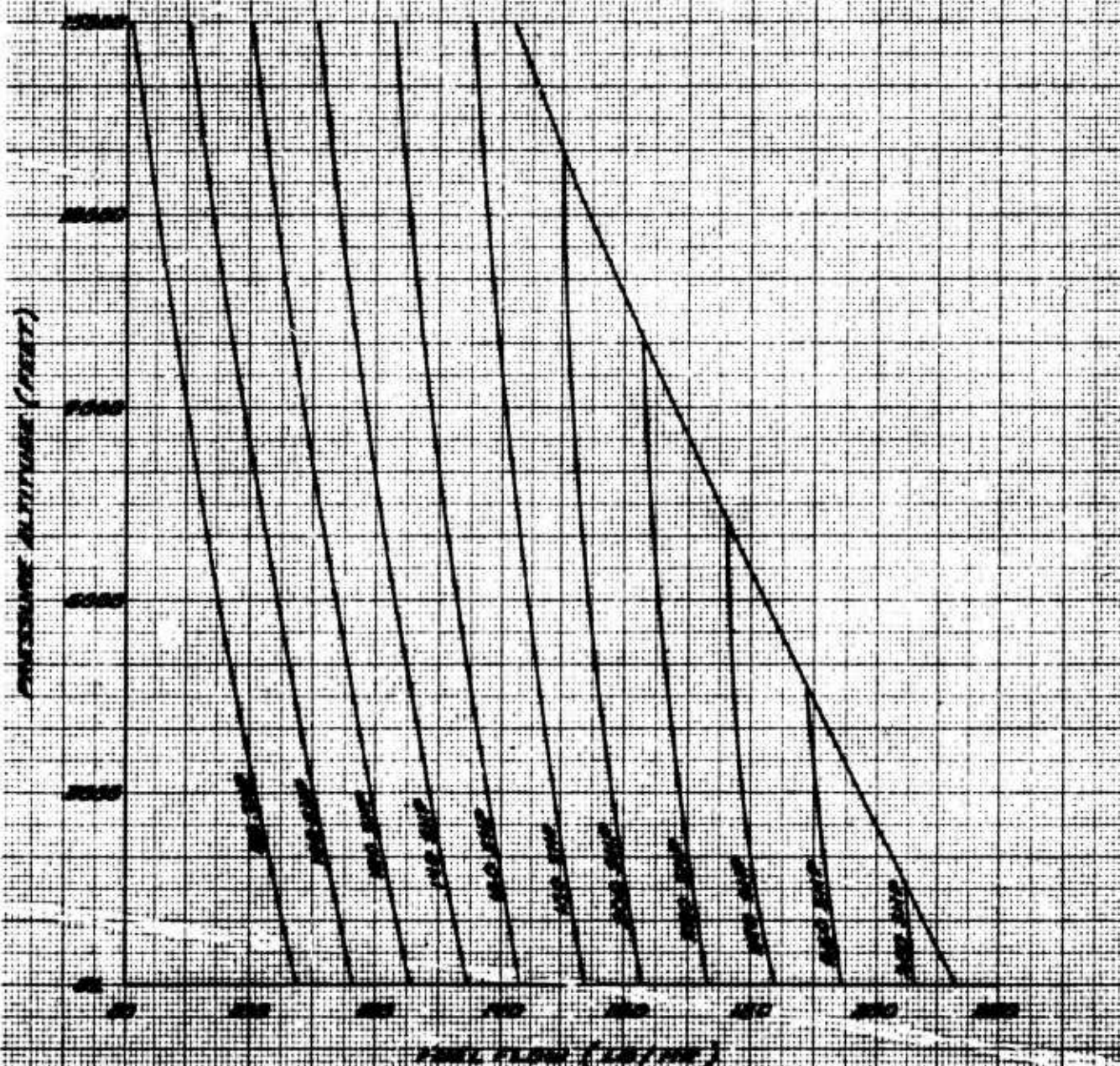


FIGURE 63  
T63-A-700 ENGINE TEMPERATURE SURVEY (°C)  
WITH HUGHES EXHAUST EXTENSIONS INSTALLED

TEMPERATURE PROBE LOCATION	FLIGHT CONDITION					500 FPM DESCENT	MAXIMUM PERMISSIBLE OPERATING TEMPERATURE
	MAXIMUM	POWER CLIMB	LEVEL FLIGHT				
---	51 (KCAS)	51 (KCAS)	52 (KCAS)	52 (KCAS)	87 (KCAS)	49 (KCAS)	---
Compressor section	76 (84)	70 (79)	69 (73)	76 (85)	75 (85)	61 (68)	135
Gearbox section	42 (46)	40 (45)	38 (44)	35 (39)	36 (41)	39 (44)	121
Turbine and combustor section	78 (86)	71 (80)	78 (89)	80 (90)	78 (88)	81 (91)	232
Top engine mount pad surface	95 (104)	85 (96)	90 (103)	85 (95)	86 (95)	90 (101)	160
Ignition harness surface	79 (87)	58 (65)	59 (68)	70 (78)	61 (69)	56 (63)	232
Thermocouple harness surface	127 (140)	112 (126)	112 (128)	120 (134)	111 (126)	118 (132)	315
(Exit)	76 (84)	76 (86)	76 (87)	75 (84)	75 (85)	75 (84)	
Oil cooler temperature	84 (92)	80 (90)	78 (89)	79 (88)	77 (87)	78 (87)	107
(Inlet)							
Maximum ambient temperature	22.5	15.0	10.5	17.5	13.5	17	52
Pressure altitude (feet)	2520	5840	7960	4320	6320	4280	---

NOTE: Values in parentheses are correct.